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CHINESE SPACE SCIENCE AND TECHNOLOGY

(Selected Articles)

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TITLE: TECHNICAL SUMMARY FOR COMMUNICATION SATELLITES

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SUMMARY This article carries out a technical summary for communication sattelites. It describes limiting conditions in the process of designing communications satellites, and it details primary technical design characteristics of sub-systems on satellites, for example, effective loading, attitudes, and orbit controls, as well as power sources, measurement control, structures, and heat control, along with other similar items.

Key Terms. Communications Satellite, Technology, Synthesis, Analysis

Since the successful launching of the first international communications satellite--Morning Bird--in 1965, satellite communications have come more and more to be the focus of world attention. There have already been 170 synchronous communications satellites launched into space. From technical investigations to practical applications, these have made satellite communications technology more familiar and more advanced with every day that passes. Since 1984, our country has launched 5 synchronous communications satellites. Technically speaking, they have been divided into two generations, causing our country's satellite communications to go from an experimental phase into a practical application phase. At the present time, our country is just in the midst of test manufacturing a third generation of high capacity communication satellites--the Red Star No.3. In spite of the fact that, according to their uses, it is possible to take communication satellites and divide them into various classes of different models, that, however, is only speaking in terms of design techniques and nothing more. They have a good number of common characteristics. This article carries out its summary aimed at these several common points of design technology.

I. CONSIDERATIONS ASSOCIATED WITH LIMITATIONS ON LARGE SYSTEM CONNECTIONS

Satellite communications engineering is a complicated aspect of systems engineering. The state of satellite technology is impacted by

the restrictions of several types of conditions. One among these is nothing else than the limitations associated with the several large system connection or interface conditions, such as, user, lifting rocket, launching site, measurement control or telemetry nets, and other similar considerations. For example, the user imposes limitations on satellite utilization requirements and effective loads. There are limitations associated with the lifting rocket lift capabilities, machinery and electrical connections, as well as the dynamics or mechanics environment. There are limitations associated with the geographical characteristics of the launch site as well as the launch facilities. And, there are limitations associated with telemetry frequencies and coverage stations, as well as other similar items. These limiting conditions are the basis for selection of satellite technology plans.

As far as the area of lifting rockets is concerned, the important consideration is the nature of the satellites they will permit one to launch and the costs of launching. At the present time, in the world, there are not very many lifting rockets capable of being used to launch synchronous satellites. Always, in order to move toward the international market, satellite designs come to be involved with a few lifting or carrier rockets. Carrier rockets that are capable of supplying launches for synchronous satellites are as shown in Table 1.

Our country's first and second generation communications satellites were launched by the Long March No.3 rocket. This type of rocket not only possesses a very high probability of success, but, possesses, moreover, a very high precision of control. Our country's third generation high capacity communications satellite is projected to be launched by the use of the modified model lifting or carrier rocket--the Long March No.3 A. Designed at the same time, it is considered to be similar to the Arian lifting or carrier rocket.

Our country's Xichang launch center is an excellent launch site for synchronous satellites. Because its latitude is relatively low, it makes the angle of incidence or inclination of orbits relatively small. As a result of this, the amount of energy needed to change the orbit of satellites is reduced. It is capable of saving 15-20% on the energy needed to change orbits as compared to the Soviet Baikenor launch site.

1 运载火箭名称	国 12 家	同步轨道有效载荷大约质量（除远地点发动机） (kg) 18	整流罩有效载荷直径 (m) 19
2 德尔他2 914或3 914	13 美国	340～460	2.2
3 阿特拉斯—人马座	13 美国	870	2.9
4 航天飞机—PAM/D	13 美国	640	2.9
5 大力神Ⅲ C/阿吉拉	13 美国	1 470～1 940	3.1
6 大力神Ⅲ E/人马座	13 美国	3 360	3.7
7 阿里安Ⅲ & Ⅳ	14 欧空局	1 500～2 470	2.9～3.0
8 N火箭	15 日本	130	1.4
9 联盟号	16 苏联	2 184	/
10 质子号	16 苏联	1 050	/
11 长征三号	17 中国	755	2.9

Table 1 Lifting or Carrier Rockets Capable of Launching Synchronous Satellites (1) Lifting or Carrier Rocket Nomenclature (2) Delta 2914 or 3914 (3) Atlas--Sagittarius (4) Space Plane--PAM/D (5) Titan III C/Aquila (6) Titan IIIE/Sagittarius (7) Arian III&IV (8) N Rocket (9) The "Union" (10) The "Proton" (11) Long March No.3 (12) Nation (13) U.S. (14) European Space Agency (15) Japan (16) U.S.S.R. (17) China (18) Approximate Mass of Effective or Useful Load for Synchronous Orbit (kg) (Except Apogee Engines) (19) Cowling Effective or Useful Load Diameter (m)

II. DESIGN OF EFFECTIVE OR USEFUL LOADS

The useful load of communications satellites includes the two parts of the transmitters and the communications antennas. As far as changes in the development of communication satellite technology are concerned, to a very large degree, they have been realized in changes in the development of effective or useful loads. The overall trends in their development are the expansion of communications capacity, increases in the numbers of transmitters, opting for the use of

multiple communications frequency band and multiple wave bundle or packet antennas, and increasing service life. For example, in the case of International VI as compared to International I, the communications capacity was raised 153 fold, the number of transmitters was increased 24 fold, and the life was expanded 6.7 fold. As a result, this made the expenses for each voice channel year decrease 163 fold. Our country's first generation communications satellites only had two transmitters with a total radiofrequency power of 16W. The second generation had four transmitters with a total radiofrequency power of 40W. The East Is Red No.3, which will be launched late in 1992, will have 24 transmitters with a total radiofrequency power of 240W.

1. Frequency Band Selection

The earliest communications frequency band that was selected for use the most was the C frequency band. The reason for this was that it was a very good window for radio communications. It was attenuated very little by the atmosphere. In conjunction with this, it was capable of hooking in for common use on ground based microwave relay systems. However, the C frequency band got more and more crowded as time went on. Following along with this, the magnitude of the satellite communications industry unceasingly increased, and high digital communications developed without a let up. This required the opening up of new frequency bands, for example, the Ku frequency band (14/12 GHz) and the Ka frequency band (30/20 GHz). Even higher frequency ranges were capable of causing earth station antennas to be even smaller, receive a high gain, narrow wave packet, and, at the same time, also be capable of receiving an even larger effective band width. For example, in the 20/30 GHz frequency band width, the upper and lower lines are capable of using operating frequency bands which each reach 2500 MHz. That is 4 fold higher operating frequency bands as compared to 4/6 GHz. The advantages and disadvantages of three types of frequency bands are compared as shown in Table 2.

Ku frequency range technology is already extremely familiar. From the early 1980s, the U.S. made use of it on the International V and commercial satellites. The Ka frequency band also began to be utilized in the middle 1980s, for example the Japanese Cherry Blossom

(Sakura Hana) No.1 and No.2 domestic communication satellites.

Even higher communications frequency bands bring with them the problems of large signal attenuation caused by the atmosphere and rainfall (the K frequency band, as compared to the C frequency band, has greater attenuation by 1-3 dB). At the same time, there is also the problem of the need to test manufacture new K frequency band equipment. Considered from the point of view of compatible adaptability and economy, the development of general frequency band communication satellites is appropriate.

Our country, at the present time, has several generations of satellites all of which are C frequency band satellites. At the same time, we are just in the midst of test producing K frequency band transmitters to act as technological preparation for an even newer generation of communication satellites.

① 频段		C		Ku		Ka	
② 频率 (GHz)		4	6	11.5	14	20	30
③ 可用带宽 (MHz) *		500	500	1 250	1 000	2 500	2 500
④ 天线峰值增益	⑥ 直径 2 m (dB)	35.5	39	44	46	49.5	53
	⑥ 直径 4 m (dB)	41.5	45	50	52	55.5	59
⑤ 点波束大小	⑥ 直径 2 m (度) ⑦	2.6	1.7	0.9	0.75	0.5	0.35
	⑥ 直径 4 m (度) ⑦	1.3	0.85	0.45	0.37	0.25	0.17

⑧ * C频段扩展到上行1 350MHz，下行1 100MHz

Table 2 A Comparison of the Advantages and Disadvantages of Three Types of Frequency Bands (1) Frequency Band (2) Frequency (3) Useable Band Width (4) Peak Value Antenna Gain (5) Point Wave Packet Size (6) Diameter (7) Degrees (8) * The C frequency band extends out to the upper line of 1350 MHz and the lower line of 1100 MHz

2. Multiple Use of Frequencies and New Forms of Antennas

As far as communication satellites in the early period were

concerned, the main obstacle to raising the communications capacity was low satellite effective radiated power. However, after large capacity communication satellites came out, the limitation moved from being one of power to being one of band width. As far as opting for the multiple use of frequencies was concerned, it expanded the true useable band width and solved the band width limitation. For example, the International V opted for the use of polarized separation and spacial separattion technologies, causing the effective band width to expand 2.3 fold. The International VI will expand the effective band width 3.5 fold.

As far as our country's previous two generations of satellites is concerned, due to the fact that the number of transmitters was small, there was no need for the multiple use of frequencies. However, in the case of the East Is Red No.3 satellite, by contrast, there is a need to opt for the use of polarized isolation in order to satisfy the need to be able to arrange 24 transmitters within a 500 MHz band width.

As far as the unceasing advances in the capabilities of communications antennas and structural forms is concerned, it has promoted an increase in the capabilities of communication satellites as a whole. In the early period, communications antennas were omnidirectional antennas. Later, directional antennas were developed as well as multiple feed source binding wave packet antennas, causing antenna gains to increase 6-8 fold. The antennas of our country's first generation of communication satellites were covered sphere wave packet antennas. The antenna gains were only 14dB. The second generation of satellites opted for the use of single ellipse wave packets covering all of China. The gains reached 23dB. The third generation of antennas were multiple feed source formed wave packet antennas. Gains reached 27dB. Their diameters were 2 m. In order to fit into the carrier cowling, they were made expandable.

3. Model Selection of Final Stage Power Amplifiers

In early satellites, final stage power amplifiers all used traveling wave tubes. After arsenic gallium field effect tubes came out, it was not until the beginning of the 1980s that utilization was begun of solid state amplifiers. Although, generally, solid state amplifier transmitter power was only half that of traveling wave tube

amplifiers, due to their light weight, however, and their good linearity, stable characteristics, long life, and other similar advantages, they were still able to receive peoples' favor, and they were selected to be low power amplifier components.

Our country's first generation of communication satellites opted for the use of fully "traveling amplification" types. The second and third generations opted for the use of mixed "traveling amplification" and "solid amplification" types.

III. SELECTION OF ATTITUDE STABILIZATION TYPES

The attitude stabilization forms for communications satellites have evolved into two branches. One branch is one which takes as its basis autorotational or spin stabilization, including single autorotational or spin stabilization, double autorotational or spin stabilization, gyroscopic body double autorotational or spin stabilization, and other similar forms. The other branch is body or cubic stabilization, that is, three dimensional stabilization. As far as several other forms of stabilization are concerned, for example, gravitational gradient stabilization, geomagnetic field stabilization, triautostabilization or triple spin are all only just in the experimental stage or only exist as a theory. They have not achieved practical operational use. The evolution of attitude stabilization is as shown in Fig.1.

As far as single autorotation or spin and the generality of double autorotation or spin systems are concerned, the satellite body must be designed so as to form a short and wide shape. Because of this, there are special requirements imposed on the instrument lay out and the diameter of the lifting or carrier rocket cowling. Gyroscopic body double autorotation or spin stabilization overcomes this type of shortcoming. However, its silicon solar battery adhesive strip surface area is limited, influencing increases in the power of 17
electric power sources. Since its development in the early 1980s, the telescopic type of gyroscopic body double autorotation or spin stabilization has resolved the problems of supplying large amounts of power from electric power sources as well as the question of fitting into the lifting or carrier rocket cowling.

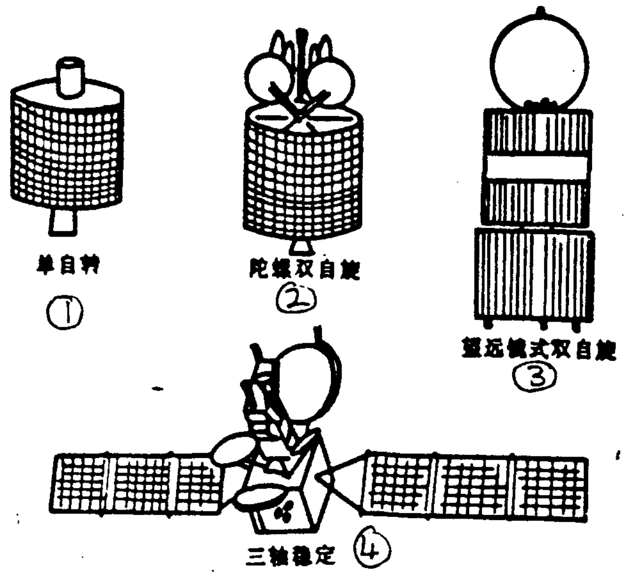

Fig.1 The Evolution of Attitude Stabilization (1) Single Autorotation or Spin (2) Gyroscopic Dual Autorotation or Spin (3) Telescopic Type Dual Autorotation or Spin (4) Three Axis Stabilization

(1) 姿态稳定方式		(10) 姿控精度	(15) 电源功率	(19) 总体布局要求**	(26) 可允许通信容量	(30) 同样功率太阳电池贴片数	(35) 技术难度	(40) 发展潜力
(2) 自旋稳定	单自旋 (4)	(11) 低	小 (16)	星本体短粗形，要求 $\mu>1$ (20)	小 (27)	需 8 倍 (31)	简单 (36)	无 (41)
	双自旋 (5)	较低 (12)	小 (16)	同　上 (21)	小 (27)	同上 (32)	同上 (37)	少许 (42)
	陀螺体双自旋 (6)	高 (13)	较大 (17)	要求绕自旋轴对称布局，$\mu<1$ (22)	较大 (28)	同上 (32)	较难 (38)	较大 (43)
	望远镜式双自旋 (7)	高 (14)	大 (18)	同　上 (23)	大 (29)	同上 (32)	难 (39)	大 (44)
(3) *三轴稳定	非全三轴 (8)	高 (14)	大 (18)	要求绕自旋轴对称布局，并要求动平衡 (24)	大 (29)	只需 1 倍 (33)	难 (39)	大 (45)
	全三轴 (9)	高 (14)	大 (18)	不要求动平衡 (25)	大 (29)	同上 (34)	难 (39)	大 (46)

(47) * 非全三轴稳定——仅同步轨道三轴稳定；
(48) 全三轴稳定——转移轨道和同步轨道都为三轴稳定。
(49) ** μ——自旋轴转动惯量与横轴转动惯量之比。

Table 3 A Comparison of Two Types of Stabilization Forms (1) Attitude Stabilization Form (2) Autorotation or Spin Stabilization (3) Three Axis Stabilization (4) Single Autorotation or Spin (5) Double Autorotation or Spin (6) Gyroscopic Body Double Autorotation or Spin (7) Telescopic Type Double Autorotation or Spin (8) Noncomplete Three Axis (9) Complete Three Axis (10) Precision of Attitude Control (11) Low (12) Relatively Low (13) High (14) High

(15) Power of Power Source (16) Small (17) Relatively Large (18) Large (19) Requirements for Overall Body Lay Out (20) The Body of the Satellite Itself Is a Short, Wide Form, Requirement > 1 (21) Same As Above (22) Requirement for a Symmetrical Lay Out Around the Axis of Autorotation or Spin, < 1 (23) Same as Above (24) Requires Symmetrical Lay Out Around Axis of Autorotation or Spin. Moreover, Requries Dynamic Equilibrium or Balance. (25) Does Not Require Dynamic Balance (26) Communications Capacity Which It Can Allow (27) Small (28) Relatively Large (29) Large (30) No. of Solar Battery Panels for the Same Power (31) Requires 3 Fold (32) Same As Above (33) Only Requires 1 Fold (34) Same As Above (35) Degree of Technical Difficulty (36) Simple (37) Same As Above (38) Relatively Difficult (39) Difficult (40) Latent Potential for Development (41) None (42) A Little (43) Relatively Large (44) Large (45) Large (46) Large (47) *Noncomplete Three Axis Stabilization--Only Synchronous Orbit Three Axis Stabilization (48) Full Three Axis Stabilization--Moving Orbits and Synchronous Orbits Both Have Three Axis Stabilization (49) ** --The Ratio Between Spin Axis Rotational Inertia and Lateral Axis Rotational Inertia.

Three axis stabilization is one new form of stabilization. It overcomes the deficiency of low power for use from solar batteries, satisfying the requirment for even larger sources of electrical power. At the same time, it also acheives even better precision of control.

A comparison of the advantages of autorotation or spin and three axis stabilization is as shown in Table 3.

IV. PROPULSION SYSTEM TECHNOLOGY

Propulsion technology and control forms are intimately related. In the early period, as far as propulsion for changing orbits on communications satellites at apogee was concerned, in all cases one opted for the use of solid engines. Attitude and orbit control made use of single component anhydrous hydrazine thrust devices. This type of propulsion system's specific impulse is relatively low. The interchangability of the system's hardware is bad. And, its control precision is relatively bad. Beginning in the middle 1970s, the new design being pushed for propulsion systems was a unified two component propulsion system. The first satellite in the world that opted for the use of a unified propulsion system was the Symphony satellite. It flew successfully, proving the superiority of unified propulsion systems. The unification of fuel used increased the rate of utilization. The engines could carry out standardization. In

conjunction with this, it was possible to control speed and shut down, increasing the precision of control. Relative impulse was high, causing the amount of fuel required to be reduced by 8-10%.

The sway of liquid fuel can bring with it difficulties for control system stability design. This is particularly true in conditions of autorotational or spin stabilization and noncomplete three axis stabilization. As far as considering the sway factor is concerned, one normally selects solid apogee propulsion adding single component anhydrous hydrazine attitude and orbit correction designs.
However, in the case of liquid fuels, two component propulsion 18
systems, it seems, are twin brothers, and are very, very commonly selected at the same time.

Our country's previous two generations of communications satellites were double autorotational or spin stabilized. In propulsion, they opted for the use of a solid added element of hydrazine. The precision of the antenna direction was only 0.6°. However, the third generation of communication satellites will opt for the use of three axis stabilization and liquid double element propulsion designs. The precision of the antenna direction will be capable of reaching pitches and rolls which are both smaller than 0.15° with flight deviations smaller than 0.5°.

V. ELECTRIC POWER SOURCE SYSTEM TECHNOLOGY

Electric power source system designs and capabilities have extremely great influences on effective communication loads. At the present time, the transmitter power consumption in high capacity communications satellites takes up approximately 80-85% of the total power comsumption of the whole satellite. Because of this, in order to reduce the weight of the whole satellite, at one time, all the electric power sources developed in the direction of high mass compared to volume and high efficiency. In order to guarantee the uninterrupted nature of communications, almost all communication satellites were designs which supplied electricity by connecting solar batteries and storage batteries.

1. Solar Batteries

Solar batteries that have practical value are N-P model slicon

batteries. In the early period, solar batteries were universal batteries. Their transfer efficiency was approximately 10%. In order to raise the efficiency, in the middle 1970s, experiments were begun with high efficiency batteries, for example, violet light batteries and nonreflective batteries have efficiencies which are capable of reaching 14-15.5%. Due to the fact that this type of high efficiency battery is expensive and irradiation losses are relatively great, as a result, operational applications were relatively rare. At the present time, there is a good deal of opting for the use of improved model universal batteries. Improved measures were shallow junctions, dense grids or rasters, back reflection bodies, or back reflection fields. In these ways, it was possible to make efficiencies increase to 12-12.5%.

Our country's first generation communications satellites selected for use universal model solar batteries. Transfer efficiencies were approximately 10%. Second generation communications satellites selected for use improved technology universal solar batteries. Their efficiency was 11.0%. Third generation communication satellites will opt for the use of shallow junction, dense grid or raster, back reflection body batteries. Efficiencies are capable of reaching 12.0%.

2. Storage Batteries

In the early period, communications satellite storage batteries were cadmium-nickel batteries. From the early 1980s, use was also begun of hydrogen-nickel batteries. If one compares these two types of batteries, although hydrogen-nickel batteries had discharges with very high voltages, discharges which are deep and large, good resistance to overcharge and overdischarge characteristics, and high mass to energy ratio, their volumes, however, were large, their structural masses were large, volumes were small compared to energy, and spontaneous discharges were large. Because of this, among those things developing storage battery technology, are the trends which exist side by side with each other toward the formation of cadmium-nickel batteries and hydrogen-cadmium batteries.

Our country's three generations of communication satellites all opted for the use of (or will opt for the use of) cadmium-nickel storage batteries. In communication satellites for specialized uses,

they will also opt for the partial use of hydrogen-nickel storage batteries.

3. Solar Battery Array Sail Panels

There are rigid, pliable, semi-rigid, semi-pliable, and other similar numerous types of solar battery array sail panel types. The vast majority of all communication satellites opt for the use of rigid sail panels. In the early period, the sail panels were honeycomb structures making use of aluminum or glass fiber reinforced plastic surface panels. Their specific power in the final phase of their lives was only approximately 10-12 W/kg. Later on, as they developed, the light model structures of relatively light density aluminum honeycomb and carbon fiber display surfaces raised their specific powers to 20-25 W/kg.

Pliable, semi-rigid, and semi-pliable sail panels are still in the experimental stage. Although their relatively large specific power values (maximums are capable of reaching 50 W/kg) are very fascinating to people, due to the fact, however, that the technology is well known and economical, in the last ten years, rigid sail panels have still been the main type.

Our country's three generations of communications satellites have had solar battery array sail panels which opted for the use rigid sail panels. These were brought in from outside the country.

VI. SELECTION OF MODEL AND LAY OUT FOR OVERALL STRUCTURE

The selections as the forms of the overall structures for autorotational or spin stabilization and three axis stabilization are entirely different. In order to guarantee symmetry around the axis of autorotation or spin, and, in conjunction with that, to cause the solar battery arrays to receive sunshine uniformly, autorotational or spin stabilization is all structured for cylindrical bodies with instruments distributed symmetrically along the axis of autorotation or spin. Three axis stabilized main body structures all opt for the use of box-shaped structures. The solar wings, at the time of lift off, are folded along the two side walls.

Following along with the requirements for serialization and development to improve models, in the last ten or twelve years, the design concept of the the common use cabin (also designated the

service cabin) has come out. This type of structural form is convenient for overall loading of experiments and changes in model. Its classic designs are the HS367 design of the Xiusi (phonetic, possibly Hughes) Company, the European Space Agency's Space Bus 100 Model, 200 Model, and 300 Model.

For the three axis stabilization satellites, there was also the development of the concept of complete modularized configurations. All that was necessary was to take the satellite and divide it into a certain number of individual modules (for example, antenna module, communications module, service module, solar wing module, propulsion module, and so on). The various modules are capable of being loaded or assembled in sets along side each other and tested. This economizes on the period involved, is convenient for overall assembly or loading, disassembly, and changes in model design.

VII. TELEMETRY SYSTEMS AND TECHNOLOGY

Since after the first synchronous satellite, Xinkang, was successfully launched, telemetry technology just became daily more familiar and finalized in design. There were 4 main tracking stations for the earliest international communications satellites. At the present time, these have already developed into 9. Among these, a good number also serve double duty as transmitters in the task of measuring orbital characteristics.

1. Telemetry Frequency Band Selection

Telemetry frequency band selection is a complicated problem. It not only requires consideration of the pass-on characteristics of telemetry equipment and its capability to be universally used internationally. It also requires simultaneous consideration of both the launch telemetry tasks of other models of satellites and current and future trends in the development of telemetry equipment. Table 4 sets out the telemetry frequency bands of several synchronous satellites.

Telemetry frequency band selection and communications frequency band selection are directly connected. In order to make comprehensive use of the hardware equipment on board satellites, for example, final stage power amplifiers used in common by communications and telemetry, telemetry beacons do double duty as communications beacons, and

telemetry frequency bands and communications frequency bands will be selected to be the same. However, frequency bands selected to be the same will also bring with them difficulties in placement, causing interference of the communications and the telemetry. In order to avoid this interference, one will always select telemetry frequencies at the two ends of the communications frequency bands.

(1) 卫星	(4) 国际通信卫星	(5) 应用技术卫星	(7) 萨康	(8) 西联星	(9) 兄弟	(10) 通信技术卫星	(11) 轨道试验卫星	(12) 同步气象卫星
(2) 国家	(6) 美国	(6) 美国	(6) 美国	(6) 美国	(13) 加拿大	(13) 加拿大	(14) 欧联	日本 (15)
(3) 测控频段	C	C	Ku	C	C&Ku	Ku	S&Ku	S

Table 4 Telemetry Frequency Bands for Several Synchronous Satellites (1) Satellite (2) Nation (3) Telemetry Frequency Bands (4) International Communications Satellite (5) Applied Technology Satellite (6) U.S. (7) Sakang (8) Xilian (phonetic, possibly Western Union) Satellite (9) Brothers (10) Communications Technology Satellite (11) Orbiting Experimental Satellite (12) Synchronous Weather Satellite (13) Canada (14) European Unicn (15) Japan

2. Telemetry Systems

What is indicated by telemetry systems is carrier wave modulation types. There are two types: frequency modulation and phase modulation. The up line is phase modulation. The down line is phase modulation and frequency modulation. Frequency modulation systems are simple. However, their precision in measuring distances is relatively low. Phase modulation systems are complicated. However, their precision in measuring distances is high. If one desires to raise the distance measurement precision of frequency modulation systems, it is necessary to opt for the use of even higher distance measuring tones and improved fixed orbit mathematical methods.

As far as the several generations of our country's communications satellites are concerned, they all opted for the use of the C frequency band to act as the telemetry frequency band. The first and second generations of communications satellites opted for the use of forms with the up line carrier wave phase modulation and frequency

modulation and the line down carrier wave phase modulation. The third generation of communication satellites had the up line with only frequency modulation and the down line phase modulation.

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VIII. HEAT CONTROL TECHNOLOGY

As far as communication satellite heat control methods are concerned, they are generally all primarily passive type heat control measures with designs for auxilliary electric heaters. However, on the basis of longer and longer lives and the characteristics of greater and greater communications capacities, there was a requirement to have the thermal control materials' attenuation factor over time small. When selecting and laying out the surface area of satellites, it is necessary to consider heat dissipation surface areas for components which produce a lot of heat. As regards three axis stabilization satellites, a number of surfaces only alternate receiving sunshine once in a period of daylight and darkness. However, another several surfaces, by contrast, will not see the sun for long periods. Because of this, heat control measures are even more complicated. In order to balance out the unequal wall temperatures of satellite bodies, it is often necessary to opt for the use of ring-shaped heat ducts as measures to conduct heat.

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SELECTION OF LAUNCH TRAJECTORIES FOR GEOSYNCHRONOUS SATELLITES

Yu Menglun

SUMMARY This article does a number of analyses on the launch trajectories or orbits of geosynchronous satellites. It discusses the relevent influences of the latitude of launch points, parking orbits, and non-coplanar launch trajectories on the selection of launch trajectories.

KEY TERMS Geosynchronous Satellite, Orbital Element, Launching, Selection

Launching geosynchronous fixed point satellites requires opting for the use of launch trajectories with multiple iterations of path alteration. In general, launch trajectories can be divided into two types. One is launch trajectories that have parking orbits. Among these, one can also divide them into the two types of parking orbits and shifted orbits that are coplanar and non-coplanar. The other is launch trajectories that have no parking orbits (for example, as shown in Fig.1).

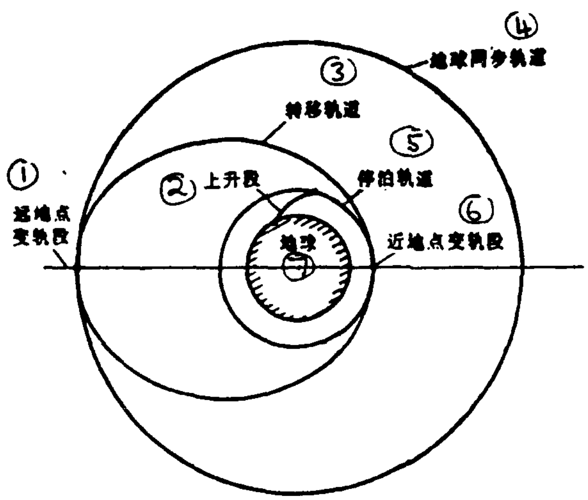

Fig.1 (a) A Schematic Diagram of Launch Trajectories that Have Parking Orbits (1) Apogee Orbital Alteration Stage (2) Ascent Stage (3) Orbital Shift Stage (4) Geosynchronous Orbit (5) Parking Orbit (6) Perigee Orbit Alteration Stage (7) Earth

Launch trajectories that have parking orbits can be divided into five sections:

(1) Ascent Stage (1st powered flight stage. Its mission is to take off from the earth's surface and make the spacecraft enter a parking orbit);

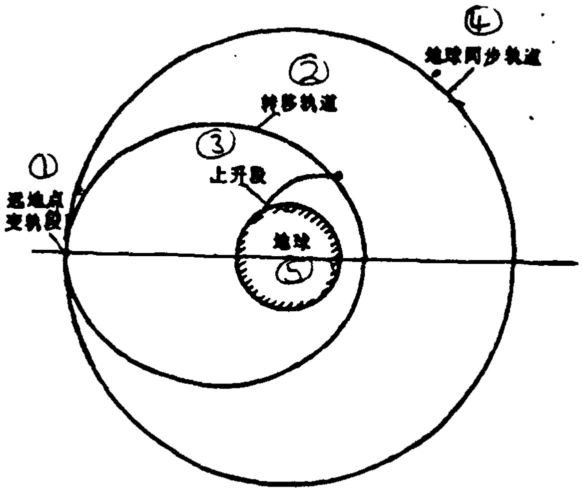

Fig.1 (b) A Schematic Diagram of Launch Trajectories Without Parking Orbits (1) Apogee Orbital Alteration Stage (2) Shifting Orbit (3) Ascent Stage (4) Geosynchronous Orbit (5) Earth

(2) Parking Orbit (Free glide or coasting stage. Its function is to adjust the postion of the spacecraft in order to insure that the main axis of the later shifting orbit is located in the plane of the equator);

(3) Perigee Orbit Alteration Phase (2d powered flight stage. Its mission is to initiate acceleration effects to cause the spacecraft to enter the perigee of the shift orbit from the parking orbit);

(4) Shift Orbit (Free glide or coast stage. Its function is to adjust the location of the spacecraft in order to guarantee that the later apogee orbit alteration enters the required geosynchronous fixed point orbit);

(5) Apogee Orbit Alteration Stage (3d powered flight stage. Its mission is, at the apogee of the shift orbit, to initiate acceleration and orbital plane alteration effects to cause the spacecraft to enter geosynchronous fixed point orbit from the shift orbit). 22

Launch trajectories that have parking orbits are utilized in areas of middle latitudes or high latitudes to launch geosynchronous fixed point satellites

Launch trajectories that do not have parking orbits are utilized in areas of low latitudes to launch geosynchronous fixed point satellites. This type of launch trajectory can be divided into three sections:

(1) The ascent stage (1st powered flight stage. Its mission is to lift off from the earth's surface and make the spacecraft enter a shifting orbit);

(2) Shifting orbit;

(3) Apogee orbit alteration stage;

Our country's Long March No.3 carrier or lifting rocket, the U.S. Sagitarius carrier or lifting rocket, and the U.S.S.R.'s Proton carrier or lifting rocket all opt for the use of launch trajectories that have parking orbits.

This article carries out analyses and discussions of a number of problems related to the selection of launch trajectories on the basis of actual experience in the design of launch trajectories.

I. THE INFLUENCE OF THE LATITUDE OF LAUNCH POINTS ON THE LAUNCHING OF FIXED POINT SATELLITES

Generally, launch sites for the launching of fixed point satellites must, as much as possible, be set in areas of low latitude near the equator. The influences of launch point latitude on the launching of geosynchronous fixed point satellites are relatively large. Principally, they are:

(1) Because of the special characteristics of geosynchronous fixed point orbits, there is a requirement for the shifting orbit's perigee argument to approach 0° or 180° (that is, the shifting orbit's primary axis located on the equator). As far as the situation

in which the launch site is positioned in areas of middle or high latitudes is concerned, generally, it is very difficult to go through a one-time acceleration to enter the shifting orbit. Moreover, it is necessary to set up a parking orbit in order to make the transition and guarantee the perigee argument of the shifting orbit. However, launching in areas close to the equator, there is then no need for parking orbits, and the lifting or carrier systems for parking orbits are complicated.

(2) The Influence of the Latitude of the Launch Point on the Required Launch Orbital Velocity. The contents which are included in the required velocity of launch orbit can be expressed using the formula below:

$$V_r = V_t + \Delta V_t + V_l - V_\omega \qquad (1)$$

In this formula, V_r is the velocity required for launch orbit. V_t is the velocity required to enter the shifting or transition orbit. ΔV_t is the velocity for the change in orbits from the shifting or transition orbit to the geosynchronous fixed point orbit. V_l is the velocity losses due to gravity, atmospheric resistance, and other similar factors. V_ω is the velocity produced by implication from the rotation of the earth. Among these, the quantities related to the latitude of the launch point are $\Delta V_{t(\text{illegible}}$ and $V_{\omega\theta}$. The formula for calculating $\Delta V_{t(\text{illegible})}$ is:

$$\Delta V_t = \sqrt{V_s^2 + V_{at}^2 - 2V_s V_{at} \cos i_t} \qquad (2)$$

In this, V_s is the velocity of the geosynchronous orbit. V_{at} is the apogee velocity of the shifting or transfer orbit. i_t is the angle of inclination of the transfer orbit. In general, the launching of geosynchronous fixed point satellites opts for the use of a direction of launch close to true east. In this way, the angle of orbital inclination $i_{t(\text{illegible})}$ for the transfer orbit is approximately equal to launch point latitude B_0. At this time, there is an approximate

$$\Delta V_1 = \sqrt{V_1^2 + V_{e1}^2 - 2V_1 V_{e1}\cos B_0} \tag{3}$$

The implied or induced velocity V_e is very difficult to use analytical formulas to express. Its precise numerical value can be obtained through solving trajectory or ballistic motion equations. To act as a qualitative analysis, it is possible to substitute using the launch point induced or implied velocity.

$$V_e = \omega_e r \cos B_0 \sin A_0 \tag{4}$$

In this formula, ω e(illegible) is the earth's angular rotation velocity. r is a geocentric radial vector. A_O is the launch azimuth or bearing angle. At the time of a true east or due east launch, $A_0=90^\circ$. At this time, $V_e=\omega_e r$ (illegible) $r\cos B_O$. The induced or implied velocity follows increases in B_O and reduces. In this way, in non-equitorial areas, as far as launching geosynchronous fixed point satellites is concerned, due to the effects which are caused by the latitude of the launch point on the required launch orbit speeds, it is possible to use the increase in velocity increment ΔV to the amount of velocity required equivalent to launching at the equator ($B_O=0$) to express them.

$$\Delta V = T + E \tag{5}$$

In this $T=\sqrt{V_1^2 + V_{e1}^2 - 2V_1 V_{e1}\cos B_0} - (V_1 - V_{e1})$ is the influence of launch latitude on orbit alteration velocities from transfer orbits to geosynchronous fixed point orbits. $E=\omega r(1-\cos B_0)$ is the influence of launching point latitude on induced or implied velocities. Results of calculations are set out in Fig.2. 23

Fig.3 is the influence of launch point latitude on the mass of the effective load or payload of a certain carrier or lifting rocket's geosynchronous fixed point orbit (in the effective load or payload mass one does not include the remaining mass of the engine at apogee).

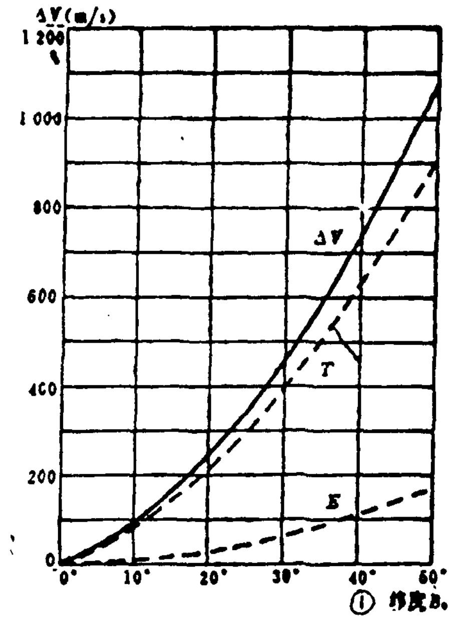

Fig.2 Effect of Launch Point Latitude on Velocity Requirements (1) Latitude

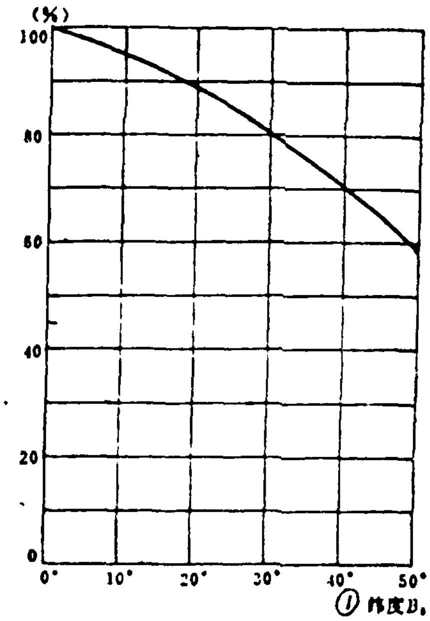

Fig.3 Influence of Launch Point Latitude on the Effective Payload in Geosynchronous Orbit (1) Latitude

II. OPTIMIZATION OF LAUNCH TRAJECTORIES WITH PARKING ORBITS

In the design stage of carrier or lifting rockets, the selection of launch trajectories must be carried out at the same time as the selection of the parameters to be put together into the overall body of the carrier or lifting rocket. Primary among the overall carrier or lifting rocket parameters that are related to the selection of launch trajectories are the propellant masses for the various stages $M_{pi(illegible)}$ and the engine thrust $T_{i(illegible)}$. Among the parameters of the launch trajectory itself are: the first stage terminal point ballistic or trajectory angle of inclination ν_{k1}; parking orbit perigee height h_{p0}; apogee height h_{a0}; perigee argument ω_0, and true perigee orbit entry point angle f_0; true perigee angle $f_{t(illegible)}$ of transfer orbit entry point, and other similar items (transfer orbit perigee height h_{pi}; apogee height $h_{at(illegible)}$; and the perigee argument $\omega_{t(illegible)}$, are given by the user). The task of launch trajectory selection is, under given restraining conditions, to select overall parameters for carrier or lifting rockets and orbit parameters which cause the effective load mass or payload to be greatest.

Launch trajectory restricting conditions include:

Sub-stage drop point range limitations $L_a \leqslant L \leqslant L_b$.

Parking orbit entry point telemetry station elevation angle limitation $E \geq E_0$

Limitations on the total mass of the carrier or lifting rocket

$$\sum_{i=1}^{N} M_i = M_0$$

Maximum velocity head limit $q_{max} \leqslant q_0$

In the formulas L_a, L_b, E_0, M_0, and q_0 are all limit or restraining values. N is the number of stages in the rocket.

As far as methods for solving optimization problems that have limits or restraints involved in them are concerned, it is possible to opt for the use of penalty function methods. Going through changes in target functions causes optimization problems that have limits or constraints in them to turn into optimization problems without limits or constraints. The target function for selecting launch trajectories 24

$$J = M_{pL} - \sum_{i=1}^{4} P_i \qquad (6)$$

In this equation, M_{pL} is the effective load mass or payload. $P_{1(\text{illegible})}$ is the penalty function for the sub-stage drop point range limit.

$$P_1 = \begin{cases} 0 & L_a \leqslant L \leqslant L_b \\ k_1[1 + a_1(L - L_a)^2] & L < L_a \\ k_1[1 + a_1(L - L_b)^2] & L > L_b \end{cases}$$

P_2 is the penalty function for the parking orbit entry point telemetry station angle of elevation limit.

$$P_2 = \begin{cases} 0 & E \geqslant E_0 \\ k_2[1 + a_2(E - E_0)^2] & E < E_0 \end{cases}$$

P_3 is the penalty function for the limit on the total mass of the carrier or lifting rocket.

$$P_3 = \begin{cases} a_3\left(M_0 - \sum_{i=1}^{N} M_i\right)^2 & \left|M_0 - \sum_{i=1}^{N} M_i\right| \leqslant \Delta M \\ b_3\left(M_0 - \sum_{i=1}^{N} M_i\right)^2 & \left|M_0 - \sum_{i=1}^{N} M_i\right| > \Delta M \end{cases}$$

P_4 is the penalty function for the maximum speed head limit.

$$P_4 = \begin{cases} 0 & q_{max} \leqslant q_0 \\ k_4[1 + a_4(q_{max} - q_0)^2] & q_{max} > q_0 \end{cases}$$

The coefficients $k_1, k_2, k_4, \alpha_1, \alpha_2, \alpha_4, a_3, b_3$, M, and other similar quantities are all constants. They must be specified on the basis of the actual situation.

As far as target functions after the introduction of penalty functions is concerned, within localized ranges, changes in function values are relatively complicated. Optimization solutions for solving this type of function can opt for the use of random testing methods. Generally, it is possible in all cases to obtain satisfactory results.

III. NON-COPLANAR LAUNCH TRAJECTORY OPTIMIZATION

In non-coplanar situations, parking orbit angles of inclination and transfer orbit angles of inclination are not equal to each other. Let i_0 be the parking orbit's orbital angle of inclination. Let i_t be the transfer orbit's orbital angle of inclination. i_s is the orbital angle of inclination for the geosynchronous fixed point orbit ($i_s \approx 0$). The parking orbit's orbital angle of inclination i_0 is basically determined by the launch latitude and the launch angle of bearing. Therefore, i_0 is precisely specified. However, the transfer orbit's orbital angle of inclination i_t is capable of being selected. The selection of non-coplanar launch trajectories is primarily the selection of transfer orbit angles of inclination. The selection of other parameters is the same as that for coplanar situations.

In order to make analytical problems convenient, assume that the alteration track or orbit between orbits is of the impulse type and that the track or orbit alteration point is on the parking orbit's descending node (Fig.4).

The ballistic or trajectory parameters for the descending node of the parking orbit are: 25

$$\begin{cases} \text{高度}^{①} & h_0 = r_0 - R \\ \text{速度}^{②} & V_0 = \sqrt{\dfrac{GM}{r_0}\left(2 - \dfrac{r_0}{a_0}\right)} \\ \text{弹道倾角}^{③} & \theta_0 = \pm\cos^{-1}\left(\dfrac{C_0}{r_0 V_0}\right) \\ \text{真近点角}^{④} & f_{e0} = 180^\circ - \omega_0 \end{cases} \tag{7}$$

[Note to (7): (1) Altitude (2) Velocity (3) Ballistic or Trajectory Angle of Inclination (4) True Anomoly or Proximate Point Angle]

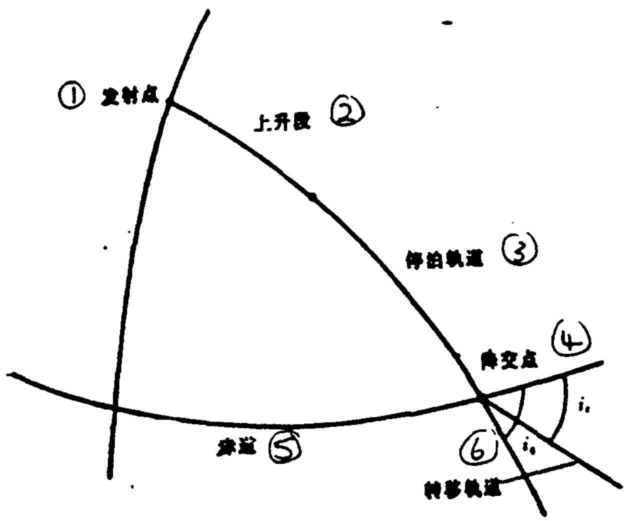

Fig.4 Non-Coplanar Orbit Alterations of Parking Orbits and Transfer Orbits (1) Launch Point (2) Ascent Stage (3) Parking Orbit (4) Descending Node (5) Equator (6) Transfer Orbits

In these equations, GM is the graviational constant of the earth. R is the earth's equatorial radius. r_0 is the geocentric vector radius. $r_0 = p_0/(1+e_0\cos f_{d(illegibel)0}$. p_0 is the through radius of the parking orbit. $p_0 = a_0(1-e_0^2$. a_0 is half the long axis of the parking orbit. e_0 is the parking orbit's eccentricity. $C_0=r_{p0}\sqrt{GM(2-r_{p0}/a_0)/r_{p0}}$. The ballistic or trajectory angle of inclination and the true anomoly have the same symbol $f_{dC(illegible)}$.

Transfer orbit ballistic or trajectory parameters at the descending node are:

$$\begin{cases} \text{高度} ① \ h_t = h_a \\ \text{速度} ② \ V_t = \sqrt{\frac{GM}{r_0}\left(2 - \frac{r_0}{a_t}\right)} \\ \text{弹道倾角} ③ \ \theta_t = \pm \cos^{-1}\left(\frac{C_1}{r_0 V_t}\right) \end{cases} \tag{8}$$

[Note to (8): (1) Altitude (2) Velocity (3) Ballistic or Trajectory Angle of Inclination]

In these equations, $a_{t(illegible)}$ is the transfer orbit half axis length. $a_t = r_a/(1+e_t)$. $r_{a(illegible)}$ is the transfer orbit apogee vector radius. $r_{a(illegible)} = h_{a(illegible)} + R$. $e_{t(illegible)}$ is the transfer orbit eccentricity. $e_{t(illegible)} = (r_{a(illegible)} - r_0)/(r_{a(illegible)} + r_0 \cos f_{t(illegible)})$. $f_{t(illegible)}$ is the transfer orbit's true anomoly at the descending node. $f_{t(illegible)} = 180° - \omega_t$ (illegible). The ballistic or trajectory angle of inclination θ_t (illegible) and $f_{(illegible)}$ are the same symbol. $C_1 = r_{at}\sqrt{GM(2 - r_{at}/a_t)/r_{at}}$.

At the descending node, one sets up the descending node coordinate system x,y,z. The coordinate origin point is at the descending node. The y axis passes through the center of the earth, pointing upward. The x axis is perpendicular to the y axis in the plane of the parking orbit and points in the direction of motion of the orbit. x,y,z is a right handed orthogonal coordinate system. The change in orbital velocity from the parking orbit to the transfer orbit, ΔV_0, has components in the descending node coordinate system which are:

$$\begin{cases} \Delta V_{0x} = V_t \cos\theta_t \cos\Delta i - V_0 \cos\theta_0 \\ \Delta V_{0y} = V_t \sin\theta_t - V_0 \sin\theta_0 \\ \Delta V_{0z} = V_t \cos\theta_t \sin\Delta i \end{cases} \tag{9}$$

In these equations, $\Delta i = i_{t(illegible)} - i_0$. This is the difference between the transfer orbit angle of inclination and the parking orbit angle of inclination.

Orbital change velocity is $\Delta V_0 = \sqrt{\Delta V_{0x}^2 + \Delta V_{0y}^2 + \Delta V_{0z}^2}$ (10)

From the transfer orbit to the geosynchronous fixed point orbit, the orbital change velocity ΔV_t(illegible) can be calculated on the basis of the formula below

$$\Delta V_t = \sqrt{V_s^2 + V_{at}^2 - 2V_s V_{at} \cos i_t} \qquad (11)$$

In this equation $V_s = \sqrt{GM/r_{at}}$; $V_{at} = \sqrt{GM(2 - r_{at}/a_t)/r_{at}}$

The total orbital change velocity (the orbital change velocity from parking orbit to geosynchronous fixed point orbit) is $\Delta V_0 + \Delta V_t$(illegible). The objective of non-coplanar orbit selection is to search out i_t(illegible) to make $\Delta V_0 + \Delta V_t$(illegible) be taken as an extremely small value. The extreme value limit is

$$\frac{\partial(\Delta V_0 + \Delta V_t)}{\partial i_t} = 0$$

After the derivation of i_t(illegible), it is possible to obtain

$$C \sin \Delta i + D \cos \Delta i = 0 \qquad (12)$$

In the equation $C = \frac{V_s V_{at}}{\Delta V_t} \cos i_t - \frac{V_0 \cos\theta_0}{\Delta V_0} \Delta V_{0z}$

$$D = \frac{V_s V_{at}}{\Delta V_t} \sin i_t + \frac{V_0 \sin\theta_0}{\Delta V_0} \Delta V_{0z}$$

In both C and D, there is the implicitly contained variable i_t(illegible). It is possible to opt for the use of numerical value methods to solve equation (12).

An actual numerical value case would be as follows. Assume the parking orbit is a 200km circular orbit. The angle of orbit inclination is 28°. The transfer orbit apogee altitude is 35,793km. The transfer orbit perigee argument is 180°. Solving for the extreme value conditions of (12), one obtains the extreme value

solutions

$i_{t(\text{illegible})} = 25.9^{\circ}$ that is, $\Delta i_{t(\text{illegible})} = -2.1^{\circ}$

The check is whether or not, with extremely small values, it is possible to calculate the second order derivative $\partial^2(\Delta V_0+\Delta V_t)/\partial i_t^2$ It is also possible to directly calculate $\Delta V_0 + \Delta V_{(\text{illegible})}$ in order to obtain verification.

Δi (°)	i_t (°)	ΔV_0 (m/s)	ΔV_t (m/s)	$\Delta V_0+\Delta V_t$ (m/s)
0	28.0	2 454.7	1 825.3	4 280.0
- 1	27.0	2 459.6	1 803.5	4 263.2
- 2	26.0	2 474.4	1 782.2	4 256.6
- 3	25.0	2 498.8	1 761.4	4 260.2
- 4	24.0	2 532.6	1 741.1	4 273.6
- 5	23.0	2 575.3	1 721.3	4 296.6
- 6	22.0	2 626.6	1 702.2	4 328.7
- 7	21.0	2 685.8	1 683.6	4 369.5

Table 1 $\Delta V_0 + \Delta V_{t(\text{illegible})}$ Changes With Δi

The calculation results described above are drawn out as the curve in Fig.5.

From the Fig. below it is possible to explain clearly why the extreme value solution $\Delta i = -2.1^{\circ}$ makes $\Delta V_0 + \Delta V_{t(\text{illegible})}$ take extremely small values.

Non-coplanar launch trajectories are advantagous to lowering orbit change velocities. However, non-coplanar changes in orbit will cause perigee orbital alteration errors to greatly increase their influence on transfer orbits. Fig.6 is the relationship between the influence of perigee orbit alteration velocity deviations on half the long or major axis of the transfer orbit and Δi.

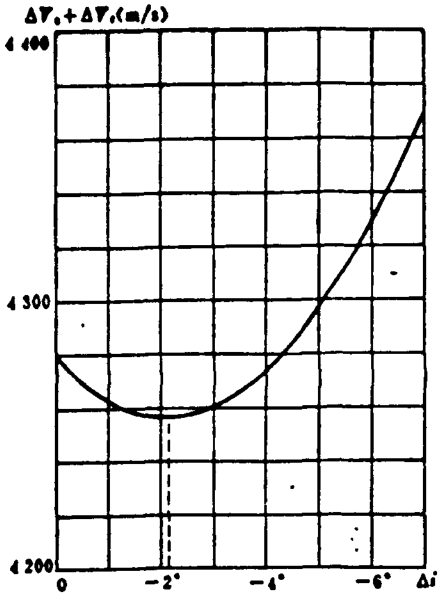

Fig.5 Changes of $\Delta V_O + \Delta V_{t}$ (illegible) With Δi

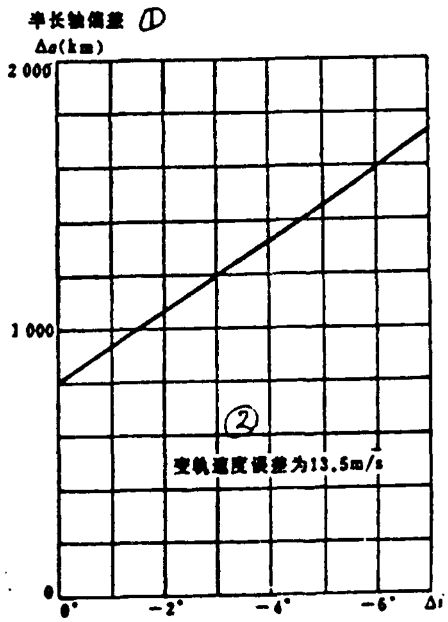

Fig.6 The Relationship Between Transfer Orbit Half Major Axis Deviations and Δi (1) Half Major Axis Deviations (2) Orbital Alteration Velocity Deviation is 13.5 m/s

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TWO TYPES OF ATTITUDE CONTROL METHODS FOR GEOSTATIONARY COMMUNICATIONS BROADCASTING SATELLITES

Lu Zhenduo

SUMMARY This article discusses the question of two types of atitude control for geosynchronous satellites which currently exist and which are just under development: satellite-ground loop control and autonomous onboard satellite attitude control.

In this article we discuss the attitude control of the STW-1 and STW-2 which our country has already successfully test produced. At the same time, it points out the past and future directions in attitude control systems on the double spin satellites our country has produced. Due to the fact that it discusses attitude control in three axis stabilized satellites, as a result, it discussed, in terms of design methods, attitude control systems for the three axis stabilized satellites which our country is in the midst of test producing.

KEY TERMS Geostationary Communications Satellite, Satellite Attitude Control System, Mathematical Model, Method.

I. FORWARD

Since the 1964 launching by the U.S. of the "Xinkang (phonetic, possible Syncom) No.3" geostationary satellite, the launching and application of geosynchronous satellites has already had 25 years of history. It has been applied in different fields, for example, communications, broadcasting, television, weather, data relay, and other similar fields. Control of this type of space vehicle is complicated. This is simply attitude control and nothing else. In transitional orbit, it is necessary to set up and stabilize ignition attitude in order to guarantee entry into the target orbit. In synchronous orbit, it is necessary to set up and stabilize directional attitude in order to guarantee the precision of the directional pointing. One must attain these requirements, and it is possible to realize them on the basis of two types of attitude control methods, that is, spin stabilization control and three axis stabilization control. As far as complete orbital spin stabilization control is

concerned, there is Intersat III and IV, as well as other similar satellites. As far as full orbit three axis stabilized systems are concerned, there are Insat, TV sat, and so on. There are also transitional orbits which are spin stabilized and synchronous orbits which are three axis stabilized for mixed control such as the Intersat V, and other similar satellites. In control forms, it is possible to opt for the use of the two types of methods, the satellite-earth loop control and onboard satellite autonomous control. Most completely spin stabilized satellites have satellite-earth loop control. Full three axis stabilization is mostly onboard satellite autonomous attitude control. Then, looking at their developmental prospects, following along with more widespread opting for the use of onboard satellite computers, onboard satellite autonomous attitude control is very much the coming thing.

II. Satellite-Earth Loop Attitude Control

1. The Concept of Satellite-Earth Loop Control

Satellite-eart' ,oop control indicates the use of equipment on the satellite and equipment on the ground mixed together to complete the tasks of the orbital control and attitude control. The equipment on the satellite includes onboard satellite telemetry and remote control systems as well as information processing and input lines, execution mechanisms, and so on. The surface equipment includes radio systems directed toward the spacecraft in order to carry out tracking and orbit measurements (for example, radar, antennas, and other similar items), the ground telemetry systems that recieve information on spacecraft attitude as well as the remote control center computers which carry out the precise determination of orbits and the precise determination of attitude, and the ground remote control systems that put out the remote control commands.

When it is necessary to adjust a certain attitude, then, on the basis of the attitude information from attitude sensor measurements on the satellite, one goes through the onboard satellite telemetry system radiating or emission antennas to broadcast it. The ground telemetry

system antennas receive it. The remote control center computers carry out precise determinations of attitude and calculate attitude adjustments. Following that, through the ground remote control system, one sends out remote control commands. Onboard the satellite, remote control systems receive and translate the codes. Through 29
onboard satellite control lines, control is exercised onboard the satellite, carrying out mechanical adjustments to the attitude of the satellite.

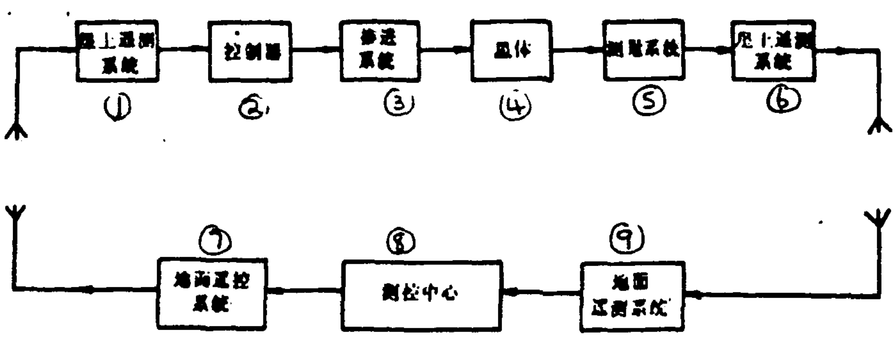

Fig.1 Diagram of the Basic Principles of Satellite-Ground Loop Control (1) Onboard Satellite Telemetry Systems (2) Control Devices (3) Propulsion Systems (4) Satellite Body (5) Measurement Systems (6) Onboard Satellite Telemetry Systems (7) Ground Remote Control Systems (8) Remote Control Center (9) Ground Telemetry Systems

2. Spin Stabilized Satellite-Ground Loop Attitude Control

Spin stabilized satellite attitude control is control of the direction of the satellite spin axis. What is called satellite-ground loop attitude control indicates opting for the use of forms of ground command control. From the ground station, through telemetry and remote control systems, ground station computers and synchronous control devices link up to form a satellite-earth loop to realize synchronous control of the satellite by the ground. Practical

execution depends on thrusting devices loaded onto the satellite in an axial direction (single component or dual component) fired in order to carry it into effect. Our country's STW-1 and 2 are nothing else but this type of control method. The engines operate according to a pulse method. There is a jet gas spray once in each spin period.

What is called attitude control software indicates, on the basis of telemetry attitude information, fuel storage pressure, as well as orbital parameters, the precise determination by the ground station of the attitude before the application of power to the craft and the attitude that needs to be established. Then, on the basis of the control rules or patterns which are selected for use, it calculates out the required number of times that a jet of gas is needed, the jet pulse length, and the phase angle of each jet of gas relative to the reference pulse, as well as other similar contents.

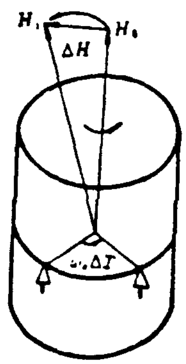

Fig.2 Schematic Diagram of Satellite Body Attitude Thrust Device Control

Thrust device nozzles are installed on the satellite body in the manner shown in Fig.2. The reaction thrust and the axis of spin are parallel. The lateral moment of control forces produced by the jets of gas is perpendicular to the axis of spin. The effects of the jets of gas cause the direction into which the angular momentum moves to lag behind the phase of the jets of gas 90°. Let the moment of force of the jets of gas be L. The spin rotation speed is ω_S. At the initial instant of initiation, the satellite is in a pure spin

configuration (generally, the gas jet moment of force is very small and it is possible to ignore nutation). As far as the approximate duration $\Delta T/2$ during the satellite's spinning to a certain phase angle is concerned, the numerical value of the increment of increase in angular momentum produced by the control of the jets of gas is equal to

$$\Delta H=\int_{-\Delta T/2}^{\Delta T/2} L\cos\omega_{1}t\,dt=L\Delta T\,\frac{\sin\left(\frac{\omega_{1}\Delta T}{2}\right)}{\frac{\omega_{1}\Delta T}{2}}$$

ΔH is perpendicular to the initial angular momentum H_O. Due to the fact that, at the time of the jets of gas, the satellite is spinning and carrying with it the moment of control forces L, it rotates in space. The angular momentum, from the initial configuration H_O, advances its movement along the circular arc to $H_{1(illegible)}$. If the gas jet effect is of a pulse type ($\Delta T \rightarrow 0$), then, the increment of increase in angular momentum is the angular momentum

$$\Delta H=L\Delta T$$

Along a straight line, from H_O, it moves in a leaping change to $H_{1(illegible)}$. Because of this, one goes through the selection of an appropriate moment for the gas jets (deciding the direction of the moment of control forces) and the duration of continuance of the gas jets (deciding the magnitude of the control impulse). After going through a number of iterations of control, it is possible to take the angular momentum and control it to any direction one desires.

The control rules or patterns are capable, on the basis of the orbital track on the celestial sphere described by the axis of spin during the process of applying power to the attitude, of being divided into the two types of great circle arc laws and equal angles of inclination laws.

(1) Great Circle Arc Method. When there is great circle arc 30
control, the track trace which is described on the celestial sphere by the spin axis is a great circle. The spin axis, in one and the same plane, is propelled from the original direction to the target attitude. This plane is fixed in space. On each iteration, the moment of lateral control forces must exist within this plane.

Because of this, when control is effected according to the great circle method, the gas jet phase is fixed in space.

(2) The Equal Angle of Inclination or Isoincline Method. At times of control, the control pulse corresponds to the attitude standard (solar or geocentric) phase angle and is a constant value in the same iteration of control. Because of this, the spin axis (the momentum moment vector quantit y) describes an isoincline arc line of forward movement in space. Isocline arc lines, although, in comparison to the great circle arc, must utilize a great deal more fuel, the control patterns or rules, on the contrary, are simple and easy to carry out. Because of this, they are often selected for use. In situations in which the initial attitude and the target attitude are already known, one, first of all, takes them and converts them to solar reference system (or earth reference system) to search out the initial attitude $\hat{P}_i$'s coordinates Ψ_i, θ_i and the target attitude $\hat{P}_f$'s coordinates Ψ_f, θ_f (See Fig.3).

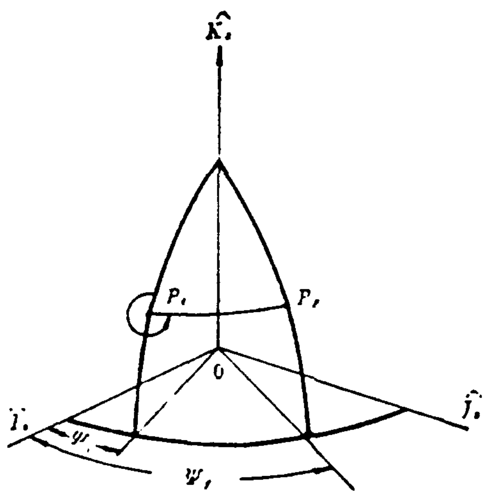

Fig.3 The Track Trace of the Spin Axis in Process of Moving Forward

On the basis of the relationships described above, one immediately solves to obtain the phase angle of the control pulse.

$$\beta_L = \begin{cases} 2\pi - \beta & \Delta\Psi > 0 \\ \beta & \Delta\Psi < 0 \end{cases} \quad (0 \leq \beta_L \leq 2\pi)$$

In the relationships

$$\beta = \mathrm{ctg}^{-1}\left\{\frac{\ln\left(\mathrm{tg}\frac{\theta_i}{2} \Big/ \mathrm{tg}\frac{\theta_f}{2}\right)}{|\Delta\Psi|}\right\}$$

If, at times when the surface synchronous control devices send out control pulses, the phase angle of their control reference standard from the surface of the earth (for example, solar standard pulses) should undergo corrections vis a vis the angles given by the equations above, when corrected, consideration should be given to nozzle installation locations, solar standard system errors and solar standard pulse width, as well as the durations of satellite-earth remote control electric signal transmissions.

The theoretical control iteration number is

$$N_t = \left[\left|\frac{S}{\Delta S}\right|\right]$$

In this, S is the control arc length from the initial attitude $\hat{P}_{i\text{(illegible)}}$ propelled to the target attitude $\hat{P}_{f\text{(illegible)}}$. ΔS is the arc length of advancing movement produced by a gas jet pulse. The symbol $[|\,|]$ expresses the selection of the values as integers or whole numbers.

Due to the changes in thrust and speed during the process of control, the true number of iterations of control should have corrections carried out on it on the basis of the theoretical number of control iterations.

3. Three Axis Stabilized Satellite-Earth Loop Attitude Control

As far as the attitude control of three axis stabilized satellites in geosynchonous orbits is concerned, ones that opt for the use of ground control are extremely few in number. Those that use it also do so, for the most part, as an emergency measure. On the basis of current understandings, Canada's domestic communications satellite, Anik-B, opts for the use of offset momentum plus magnetic control as well as single element hydrazene systems. When its east-west position is maintained, attitude control uses ground controls.

As regards autonomous attitude control systems on three axis stabilized satellites, one should consider spare or backup means which possess ground controls.

III. Autonomous Attitude Controls On Board Satellites

1. Autonomous Attitude Controls for Spinning Satellites

As regards spin stabilized satellites, one of the key questions in autonomous attitude control is the precise specification of autonomous attitudes, that is, methods making use of attitude control processors on board the satellite to take real time attitude sensor pulse data numbers and turn them into duration intervals. These data are used in order to carry out precise autonomous specifications of attitude in orbit, and, on the basis of given attitudes and the precise specifications of attitudes, in order to realize autonomous spin axis attitude control. In order to guarantee the degree of precision required in attitude control, before the attitude control, it is necessary to carry out nutation damping control. As regards the long slender bodies of spinning satellites, one generally selects active nutation damping control. In transition orbits, one uses thrusting devices to carry out nutation control. In synchronous orbits, one opts for the use of spin elimination driving nutation damping control. In both of the two cases, the nutation sensitive components are nutation acceleration meters.

Functions completed by attitude control processing circuits are:

(1) Attitude data processing;

(2) Platform spin elimination control;

(3) Control of the antenna pointing structure;

(4) Thrust device nutation control (TANC);

(5) Driven anti-spin nutation damping; (DANVD)

(6) Attitude control thrusting device coupling circuits;

(7) Spining body rotation speed control;

(8) Malfunction identification and safety circuits:

(9) Telemetry and remote control coupling or in-porting circuits.

These kinds of control types described above are also the directions in which our country's current and future test manufactured long thin bodied dual rotation attitude stabilized control systems are going (as shown in Fig.4).

In order to raise the reliability of autonomous attitude control systems on board dual rotation satellites, it is still necessary to consider the satellite-earth loop attitude control as being auxilliary.

2. Autonomous Attitude Control On Board Three Axis Stabilized Satellites

Three axis stabilized satellites generally have two kinds of stabilization types. One is such that the transition orbit is spin stabilized, and the synchronous orbit is three axis stabilized. The other kind is the type which is entirely three axis stabilized. This is also the direction of current development. This type of stabilized attitude control system is an onboard autonomous control closed-circuit system. During satellite motions, in going from one type of operational mode into another type of operational mode, many are carried out automatically. The ground only initiates and monitors effects. The main operational modes are: solar capture, earth capture, four stage ignition attitude stabilization, normal motion position maintenance, and so on.

Many satellites of this type are high capacity communications broadcast satellites. Their sail panels, antennas, and other similar components are relatively large. The storage tanks for liquid fuel

are, for the most part, over half the lift off weight. As a result of this, the mechanics are relatively complicated. It is not only necessary to consider the flexibility of components. It is also necessary to consider the shaking movements of the liquid in the fuel tanks, giving attitude control systems relatively large associated difficulties. Our country, at the present moment, is just in the midst of test producing a three axis stabilized communications broadcast satellite which is nothing else than the control type on this kind of model.

(1) Mathematical Model

Rigid Portion

$$I\ddot{\vec{r}} + B^T\ddot{\vec{q}} + C^T\ddot{\vec{p}} = \vec{\Psi}_A'\vec{f},$$

Flexible Component Portion

$$B\ddot{\vec{r}} + A\ddot{\vec{q}} + D\dot{\vec{q}} + E\vec{q} = \vec{\Psi}_A'\vec{f},$$

Liquid Fuel Shaking Movement Portion

$$Q\ddot{\vec{r}} + F\ddot{\vec{p}} + G\dot{\vec{p}} + N\vec{p} = \vec{\Psi}_A'\vec{f},$$

32

Measurements

$$\vec{m} = \vec{\Psi}_{,}'\vec{r} + \vec{\Psi}_{,}'\vec{q} + \vec{\Psi}_{,}'\vec{p}$$

In the equations

- $\vec{r}$(6xl)--rigid coordinates
- $\vec{q}$(nxl)--flexible modality coordinates
- $\vec{p}$(nxl)--shaking modality coordinates
- $\vec{m}$(mxl)--measurements
- $\vec{f}$, (6xl)--force/moment of forces transfered into the rigid body
- $\vec{f}_q$(nxl)--internal forces/moments of force transfered into flexible structures

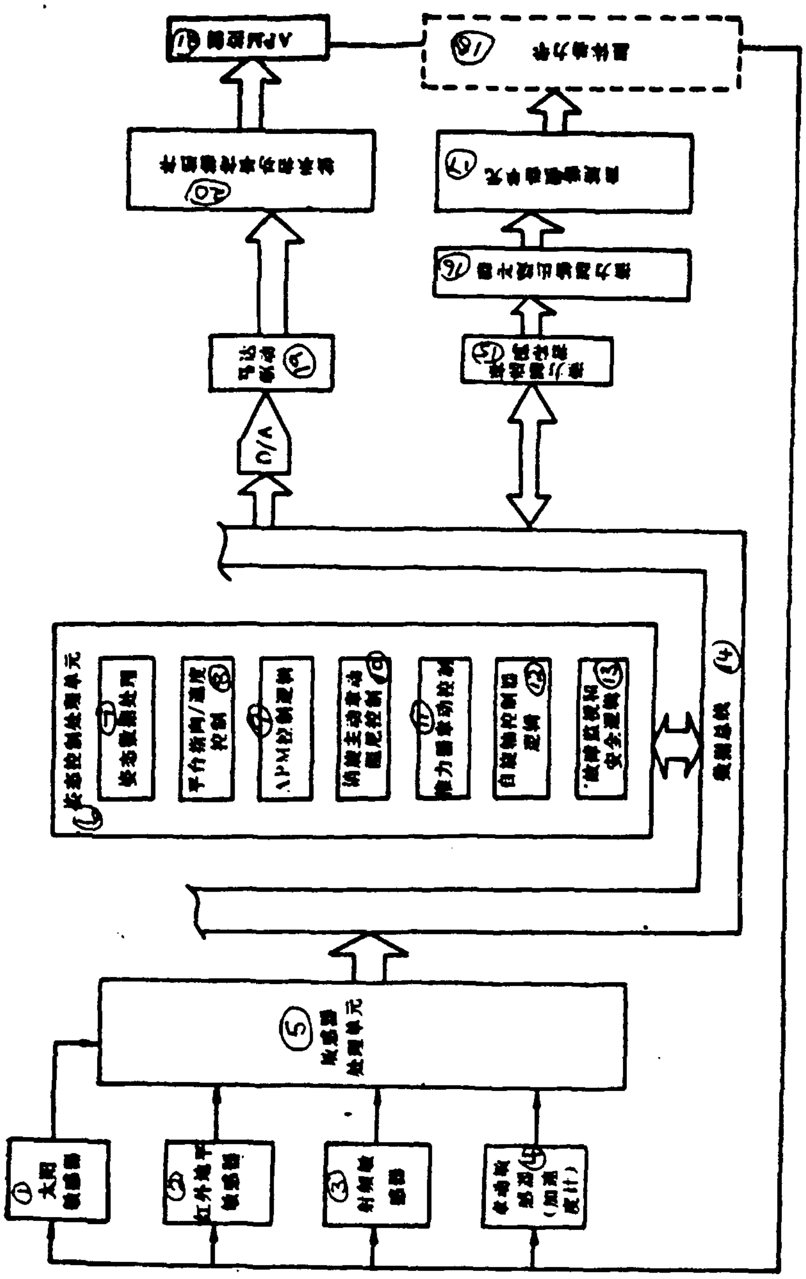

Fig.4 A Schematic Diagram of the Principles of Autonomous Attitude Control in Synchronous Orbits of Double Spin Satellites With Long Slender Bodies (1) Solar Sensors (2) Infrared Horizon Sensor (3) Radio Frequency Sensor (4) Nutation Sensor (Acceleration Meter) (5) Sensor Processing Unit (6) Attitude Control Processing Unit (7) Attitude Data Processing (8) Direction of Platform Orientation/Speed Control (9) APM Control Logic (10) Spin Elimination Driven Nutation Damping Control (11) Thrust Device Nutation Control (12) Spin Axis Control Device Logic (13) Malfunction Monitoring and Safety Logic (14) Main Data Line (15) Thruster Selection and Code Interpretation (16) Thruster Output Buffer or Bumper Device (17) Spin Axis Drive Unit (18) Satellite Body Mechanics (19) Motor Drive (20) Shaft Bearing and Power Transmission Assembly (21) APM Control

$\vec{f}_p$(nx1)--internal forces/monents of force transfered in as initiated by liquid shaking

I(6x6)--mass and inertia of rigid body

A,D,E, (nxn)--flexible component mass/damping/ rigidity diagonal line matrix

F,G,N (nxn)--mass/damping/rigidity diagonal matrix for fluid shaking of fuel in storage tanks

B (nx6)--coupling system matrix

Q,C, (nx6)--coupling system matrix

$\vec{\Psi}^r{}_A$(6x6)--rigid body characteristic vector matrix for executing structures

$\vec{\Psi}^q{}_A$(nxn)--flexible component characteristic vector matrix for executing structures

$\vec{\Psi}^p{}_A$(nxn)--fluid shaking characteristic vector matrix for executing structures

$\vec{\Psi}^r{}_s$(mx6)--rigid body characteristic vector matrix for sensor components

$\vec{\Psi}^q{}_s$(mxn)--flexible component characteristic vector matrix for sensor components

$\vec{\Psi}^p{}_s$(mxn)--shaking fluid characteristic vector matrix for sensor components

The mechanics model described above is capable of applying Laplace transforms, transforming into the frequency range and using a transmission or transfer function in order to express it:

$$\vec{m} = \vec{S}(s) \cdot \vec{f}$$

The transfer functions for sensor devices and executing structural components are

$$\vec{Y} = \vec{F}_s(s) \cdot \vec{m}$$
$$\vec{f} = \vec{F}_A(s) \cdot \vec{u}$$

In these equations $\vec{Y}$(mxl)--sensor device output

$\vec{u}$(pxl)--executing structure input

$\vec{F}_s$(s)(mxm)--sensor device transfer function matrix

$\vec{F}_A$(s)(pxp)--executing structure transfer function matrix

$\vec{Y}$ represents all controled quantities. $\vec{u}$ represents all control device outputs.

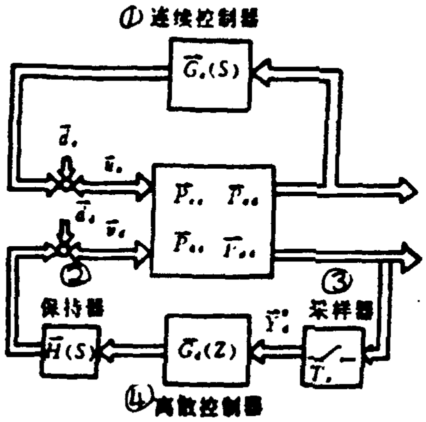

Fig.5 Control Structure Diagram of a Mixed Continuous-Dispersed Control System. $\vec{d}_c$ $\vec{d}_d$ Are Interference Inputs. (1) Continuous Control Device (2) Sustainer or Retainer Device (3) Sampling Device (4) Dispersed Control Device

In this way, a satellite mechanics model that carries in it flexible components and fuel storage tank liquid shaking movements is capable of being written in the form below.

$$\vec{Y} = \vec{P}(s)\cdot\vec{u}$$

In this equation $$\vec{P}(s) = \vec{F}_s(s)\cdot\vec{S}(s)\cdot\vec{F}_A(s)$$

(2) Analysis Design

In this way, a multiple input, multiple output complicated system, overall, is capable of being analyzed and understood as a system of two components, a continuous one and a dispersed one (as shown in Fig.5).

$$\begin{bmatrix} \vec{Y}_c \\ \vec{Y}_d \end{bmatrix} = \vec{P}(s) \begin{bmatrix} \vec{u}_c \\ \vec{u}_d \end{bmatrix} = \begin{bmatrix} \vec{P}_{cc}(s) & \vec{P}_{cd}(s) \\ \vec{P}_{dc}(s) & \vec{P}_{dd}(s) \end{bmatrix} \begin{bmatrix} \vec{u}_c \\ \vec{u}_d \end{bmatrix}$$

In these equations the subscript c indicates the continuous section
the subscript d indicates the dispersed section.

As far as stability analyses of satellites possessing non-linear control systems are concerned, generally, one opts for the use of descriptive function methods. Finally, one draws out Nyquist graphs (amplitude and phase characteristics) in order to make determinations.

On the basis of actual experience, on the amplitude and phase characteristics graphs, stability criteriia are capable of being induced to be three stability zones:

a) Phase Lead Stability Zone

In the interval between phase -180° and $+180^{\circ}$, there is appropriate stability stored up. Moreover, the gain is larger than zero decibels.

b) Phase Lag Stability Zone

In the interval between -540° and -180°, there is appropriate stability stored up. Moreover, the gain is larger than zero decibels.

c) Gain Stability Zone

The gain is smaller than zero decibels. There are no phase conditions.

In system analysis design, it is necessary to consider only coupling problems between system control frequencies based on rigid satellite body designs and vibration and shaking motion frequencies associated with first degree modality shaking movements from sail panel flexibility and fuel storage tank liquids. The overall match up of calibration grids should cause system control frequencies to be far from the frequencies of the latter two and improve the dynamic capabilities of the system.

As far as the final determination of the stability and function indicators of satellite control systems is concerned, one should utilize real time mathematical simulations and semi-physical simulations in order to complete it.

For a classical control circuit (see Fig.6).

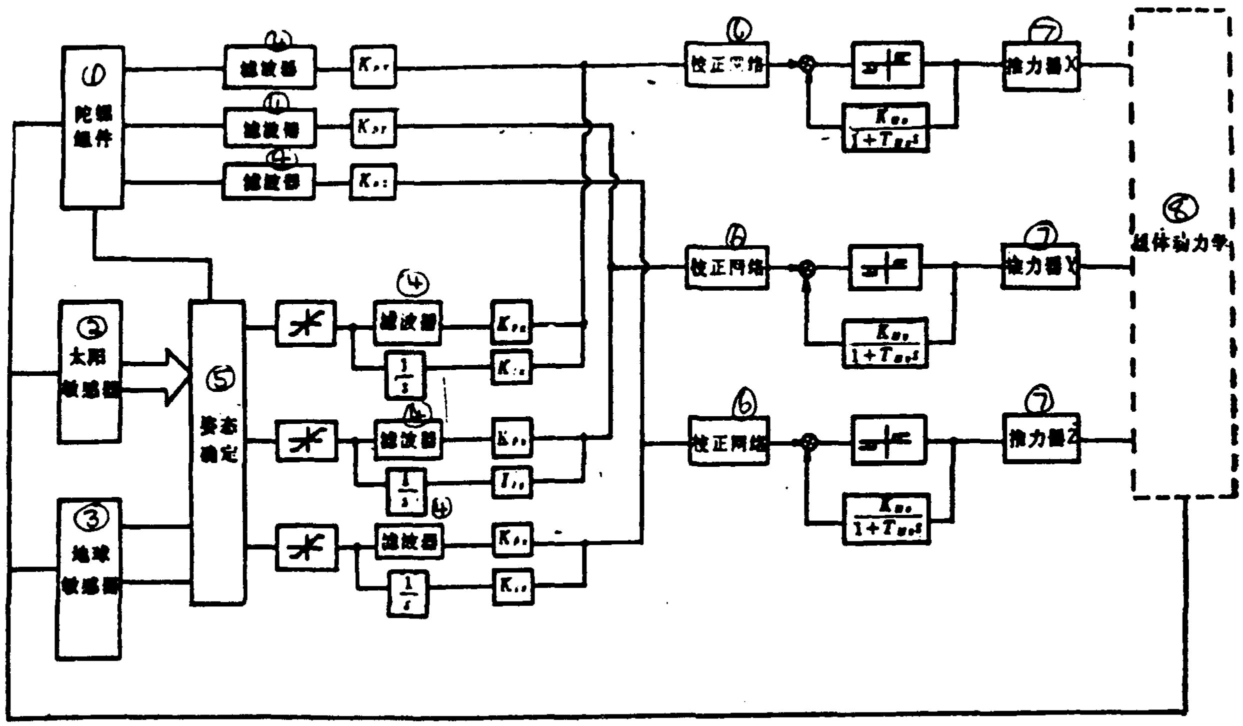

Fig.6 Distant Point Orbit Alteration and Position Maintaining Stability Circuit (1) Gyroscope Assembly (2) Solar Sensor (3) Earth Sensor (4) Wave Filter (5) Precise Attitude Specification (6) Calibration Grid or Network (7) Thruster Device (8) Celestial Mechanics

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ELECTRICAL DESIGN OF ANTENNA SUBSYSTEMS ON OUR COUNTRY'S C WAVEBAND HIGH CAPACITY COMMUNICATIONS SATELLITES

Guo Wenjia

SUMMARY This article starts from the design boundary conditions for C waveband high capacity communications satellite subsystems, and presents the antenna wave packet coverage diagram design principles which are capable of giving consideration to both of two different fixed point locations as well as methods of solution for problems encountered in designs. In particular, it concentrates special emphasis on the discussion of problems associated with wave packet formation grids or networks. Finally, it carries out an optimization design for the antenna subsystem of our country's second fixed point location communications satellite, predicting that the second satellite will be capable of achieving its targets.

KEY TERMS Communications Satellite, Antenna System, Optimization, Design

I. C WAVEBAND COMMUNICATIONS SATELLITE ANTENNA SUBSYSTEM DESIGN BOUNDARY CONDITIONS

1. Basic Design Requirements:

In order to satisfy our country's daily increasing communications and television broadcasting requirements, consideration must be given to the current state of our nation's technology, and we have decided to test manufacture a C waveband high capacity communications satellite. It requires an effective load to supply 24 individual transmiting devices operating simultaneously. As far as its coverage capabilities over our national territory are concerned, the television broadcast communications EIRP $\geq$ 38 dBW, makes the large number of individual television receiving stations in our country all capable of using a small diameter antenna (under 3 meters) in order to carry out the receiving of television signals and, in conjunction with that, make differential transmissions. The communications channel EIRP $\geq$ 35 dBW makes it possible for all earth stations to use 6 meter diameter antennas to carry out reception and transmission.

2. Plans Capable of Providing Options

On the basis of the basic requirements given in Part 1, a comparison was carried out of the various types of feasible plans, considering a compromise of numerous individual factors and optimized results, precisely specifying the design principles set out below.

(1) According to the requirements for service area coverage as well as the requirements for a compatible delivery capablility, the communications antenna diameter selected for use was a 2 meter offset parabolic surface or parabaloid antenna.

(2) In order to satisfy the service area requirement for EIRP coverage, it is necessary to select for use wave packet antennas of assigned shape to make the forms of these wave packets as much as possible match the shape of the service area in order to guarantee as much as possible that energy is concentrated in the service area.

(3) In order to adjust to the requirement for the simultaneous operation of 24 transmiting devices as well as the requirement for a high degree of separation of intersecting polarity power levels, it is necessary to opt for the use of polarized diversity and multiple frequency use techniques, that is, to opt for the use of polarity sensitive dual raster parabolic emission devices.

(4) In order to satisfy the requirement for tne simultaneous operation of 12 transmitters at each polarity within a 500 MHz band width, and, in conditions in which microwave output multiplex devices are not able to realize continuous multiplex designs, wave packet formed grids or networks cannot help but opt for the use of odd even synthesizer device designs in order to facilitate taking odd channel transmitter set signals and even channel transmitter set signals and simultaneously feeding them into antenna wave packets to form grids or networks. Besides, the whole wave packet formed grid or network ought to possess the special characteristics of broad 37
bandwidth, low comsumption, and compact geometrical dimensions, as well as other similar characteristics.

To summarize what was described above, it is possible to see that this type of C wave band high capacity communications satellite levies very great demands on effective load or payload. The difficulty of its design has exceeded the communications satellites which our country has already test manufactured in the past.

II. ELECTRICAL DESIGN OF COMMUNICATIONS ANTENNAS

1. Opting for the Use of Multiple Feed Source Offset Parabaloid Realization of Shaped Wave Packets

Offset parabaloid surface antennas possess such special characteristics as no aperture diameter sheltering, structural simplicity, and a total antenna system which is easy to draw in. Beginning in 1980, it has been widely used in satellite multiple feed source shaped wave packet antenna systems[1].

What are refered to as multiple feed source shaped wave packets are nothing else than a number of secondary wave packets produced from multiple feed sources shining on a parabolc reflecting surface and arranged within the service area. After that, one uses a certain appropriate amplitude and phase parameter to eliminate the excitation of these secondary wave packets, obtaining a wave packet form which is capable of matching up with the shape of the service area. Speaking from the point of view of physics, the more narrow the secondary wave packets which are selected for use are the more capable the wave forms obtained are of matching up with the shape of the service area. However, in terms of the practical realization of the engineering, due to the fact that there are limitations imposed by the frequencies used, the volume and weight of the antenna which the satellite permits to be installed, and other similar factors, we are not able to make each secondary wave packet very narrow. For example, speaking in terms of the C wave band, the width of the secondary 3 dB wave bundle produced with a 2 m aperture diameter antenna was approximately 2.6 degrees.

Fig.1 gives the shapes of the service areas for four different satellite fixed point locations--125°, 115°, 103°, and 87.5°. From these four range diagrams it is possible to see that our country's territory is approximately 7° in an east-west direction and 4° in a north-south direction. Because of this, it is possible to know that, for the C wave band, speaking in terms of 2 m aperture diameter antennas, it is possible to cover our country's territory in an east-west direction with four of the secondary wave packets they produce. In the north-south direction, two to three cover it. The total number of secondary wave packets was approximately seven to eight to be appropriate. According to the requirements of

optimization design, each feed source's aperture diameter or say spacing is generally selected for use as 1.1 wavelengths to 1.5 wavelengths to be relatively appropriate. The size of the aperture diameter dimensions (spacing) will produce relatively severe mutual coupling influences. Moreover, too large a feed source spacing will produce array grid segments, loss of energy, as well as, the synthesis, in wave packets, of the appearance of power level depression regions. However, the C wave band communications antennas which we currently design pay attention to both 4/6 GHz, and give consideration to the priority of guaranteeing the requirements of lower line coverage. The only thing to be done is to select feed source aperture diameter dimensions which, for the lower line, have a lowest frequency that is 1.1 wavelengths. In this way, speaking in terms of the highest frequency for the upper line, the aperture diameter dimensions, compared to wavelength, reach 1.9.

2. Realizing the Capability for One Satellite Operating in Two Different Orbital Positions Simultaneously

From Fig.1 it is possible for us to see that, in different orbital positions, the changes in service area configuration are very large. If one makes use of a $4^{O}x7^{O}$ elliptical coverage on the service areas of these several positions, generally speaking, in all cases, it is possible to take the service areas and include them inside this ellipse. In order to raise the EIRP value covering the service area, if one selects for use a shaped wave packet in order to give coverage which fits the configuration of the service area, then, at a fixed point location, when the coverage is best, a wave packet of the same type, when moved to a different fixed point location, will provide coverage which will turn very bad. The farther apart the two locations are from each other, the more difficult it will be to look after both of them. As far as optimization of coverage of the service area is concerned, there are two factors on which it depends. The first is an optimum arrangement of locations for the secondary wave packets in the service area. The second is that each secondary wave packet's excitation amplitude and phase reaches optimum values. Because of this, to make one satellite simultaneously operate in different orbital positions, one must first of all find a type of

secondary wave packet arrangement which makes it feasible for the two different orbital locations. In this way, when use is made of different orbital locations, it is only necessary to change the excitation amplitude and phase for the secondary wave packets. It is then possible to operate. This type of technique is also called wave packet reformation technology. On the premise that there is no increase in surplus wave packets, it is possible to feasiblly move the satellite's optimum fixed point location approximately $\pm 12^{\circ}$. Fig.2 gives an example. Taking 115° as the center, it is possible to look both east to 125° and west to 103° with wave packets arranged within the range in the diagram. Appropriate changes in amplitude and phase make it possible to obtain, at different fixed point locations, coverage of the service area. Fig.3 gives a coverage diagram for the three locations mentioned above as obtained under a dual modality operating configuration for a given single mode.

In order to reduce the costs for the test manufacturing of antennas onboard satellites, to make the alteration of the amplitude and phase of the various secondary wave packets as easy as possible to realize, it is possible to opt for the use of a method in which phase adjustment devices are added between two 3 dB electrical bridges in order to realize alterations in power distribution ratios. When satellites are required to operate at two different fixed point 38
locations, it is only necessary to take wave packets and form, in a grid or network, various phase shift devices to do readjustments. Then, it is possible to obtain a new set of secondary wave packet excitation parameters. If, onboard the satellites, one opts for the use of electrically adjusted phase shift devices, then, by contrast, it is possible, from the ground, through remote control signals, to carry out phase control, reaching the objective of altered wave packet configuration. The drawbacks in this type of area are that, in wave packet configuration grids or networks, one must opt for the use of relatively numerous 3 dB electric bridges. These make the grid or network change into a relatively complex one. In conjunction with this, dissipation is increased greatly. However, after opting for the use of microwave passive component integration technology, this type of grid or network, in terms of engineering, is capable of

realization. Besides this, it is necessary to deal with the fact that one satellite, when it is utilized in different fixed point locations, will have different temperature environments to handle with its antenna subsystems. Because of this, in the areas of temperature control and thermal deformations, in all cases, it is necessary to give comprehensive consideration to this.

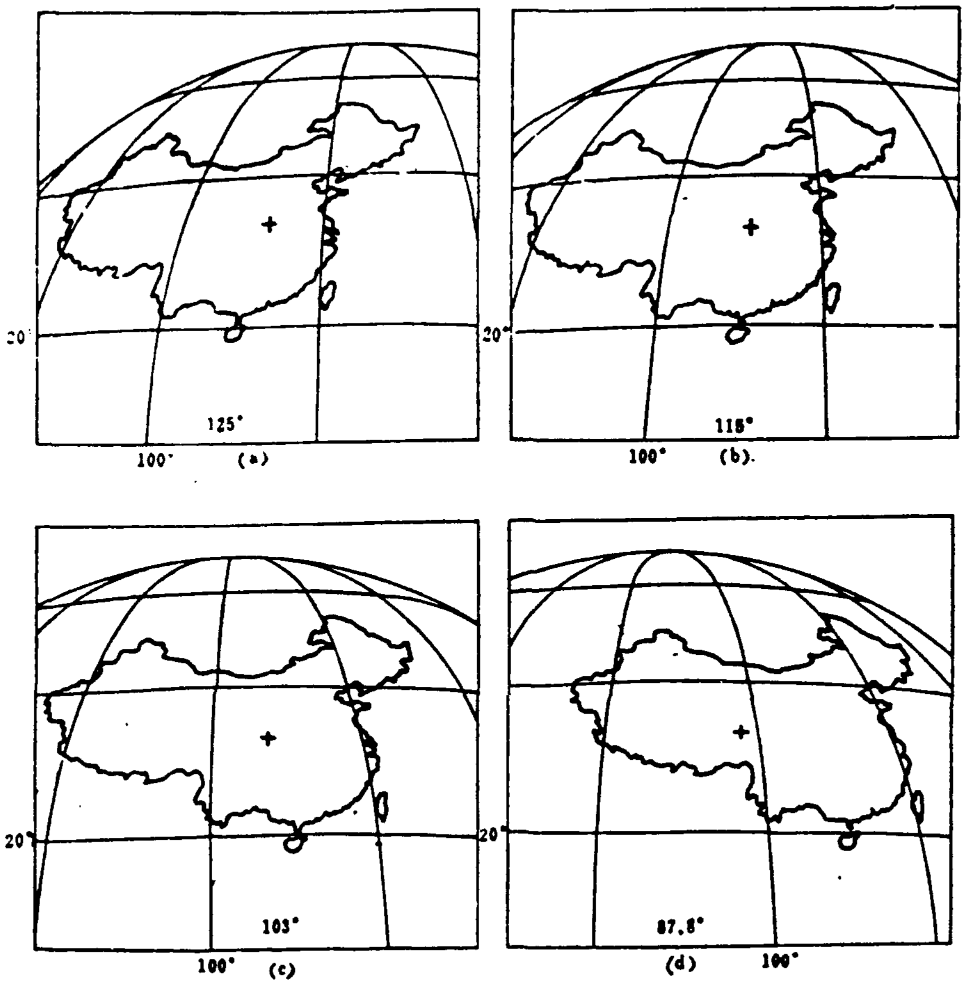

Fig.1 Service Area Configurations for Four Fixed Point Locations at 125°, 115°, 103°, and 87.5°

3. Integrated Technology for Microwave Passive Components in Wave Packet Formed Grids or Networks

Wave packet formed grids or networks are key components in the realization of multiple feed source shaped wave packet antennas. Following along with increases in wave packet shaped antenna function requirements, the number of secondary wave packets increases. In the

past, in wave packet formed grids or networks, one opted for the use of separated, individual microwave components in order to form wave packet shaped grids or networks which appeared, in all cases, in terms of weight, volume, reliability, grid or network function match up, as well as precision of structural installation, not to be able to satisfy requirements. From the middle 1980's onward, one saw the appearance of integrated technology in wave packet formed grids or networks. The classic example was that, from the beginning of the international communications satellite No. V's onboard antennas, in its wave packet formed grid or network, designers opted for the use of surface model dielectric support air belt condition curve integrated circuits, making a totally immense, complicated wave packet formed grid or network integrated circuit on one planar surface. Each feed source trumpet was directly installed on the planar surface. In this way, they made a grid or network which was extremely light in weight
and reduced in volume. Because of this, later, one saw the gradual 39
appearance, in the C wave band, of technology which opted for the use of square coaxial components to form a planar surface integrated grid or network. Due to computer assisted design and computer controled processing, it is possible to guarantee the realization of this type of complicated integrated circuit grid or network. In this C wave band high capacity communications satellite which we are currently studying, in order to reduce the antenna subsystem's weight and volume, we also opted for the use of this type of advanced technology.

In the planned operating requirement for each polarity to be able to satisfy 12 transmitters simultaneously, in order to resolve the current difficulties with the area of realizing a continuous microwave output multiplexer, in wave formed grids or networks, we added in odd-even synthesizers. Odd-even channel synthsizers have two input terminals. One terminal is directly connected to the output multiplexer of odd channel transmitters. The other terminal is connected with the even channels. Its output terminal is capable of being two terminals, three terminals, or four terminals, etc. However, in order to satisfy wave packet configuration coincidence

produced by odd and even channels, as well as guaranteeing the separation between these two channels, action as odd-even synthesizer must necessarily satisfy the relationships below:

$$\langle T^o \cdot T^e \rangle = 0 \qquad (1)$$

$$T^o = (T^e)^* \angle \phi \qquad (2)$$

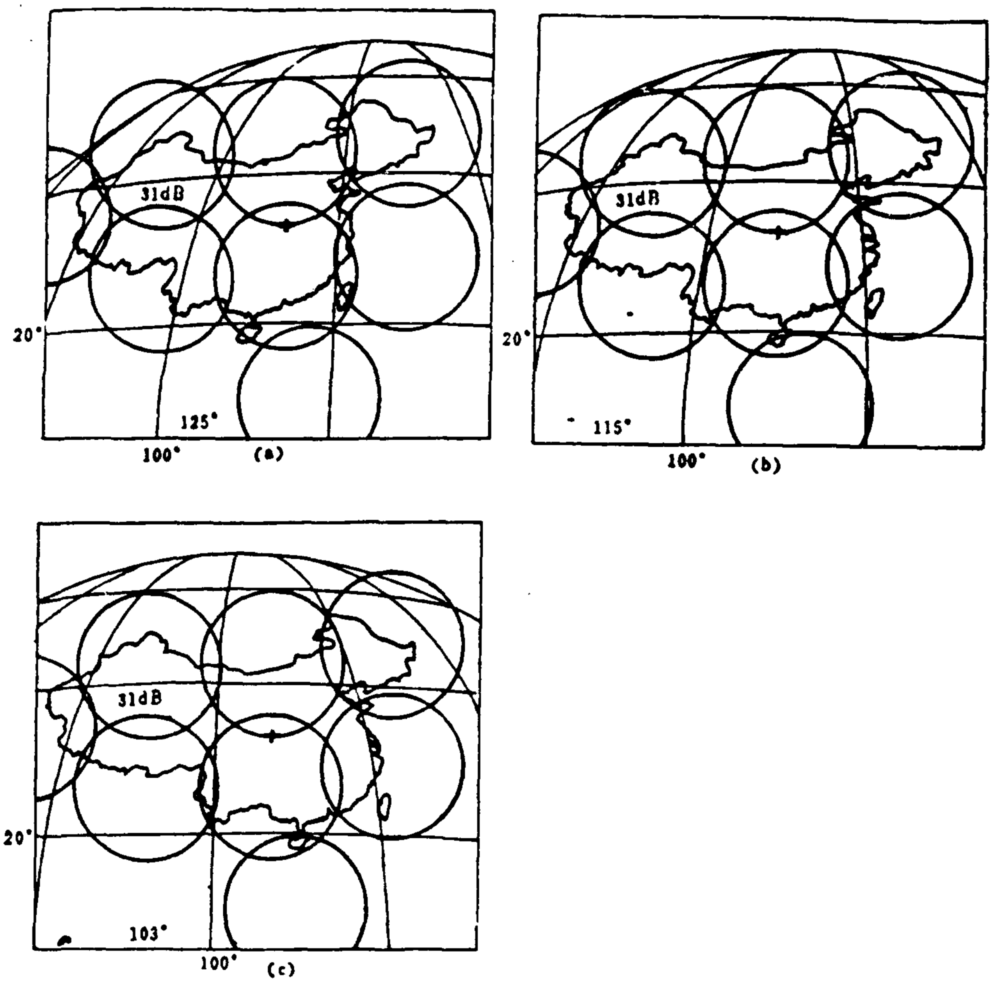

Fig.2 Diagram of Secondary Wave Packet Arrangement for Three Fixed Point Locations at 125^o, 115^o, and 103^o

T^o represents the transmission coefficients from odd terminals inputed into the various output terminals T^o_1, $T^o_2 \ldots T^o_N$.

T^e represents the transmission coefficients inputed from even terminals into the various output terminals T^e_1, $T^e_2 \ldots T^e_N$.

The first equation expresses the fact that the conjugate product of odd modality transmission coefficients and even modality transmission coefficients must be equal to zero. The second equation expresses the fact that, speaking in terms of the two modalities, their various terminal transmission coefficients, in terms of amplitude, correspond and are equal to each other. In terms of phase, they are each selected as conjugates and, after that, the difference between them is a constant phase. Odd-even synthesizers that satisfy the conditions described above are orthogonal. They make the two modalities of wave packet configuration to be maintained, basically, close to one another.

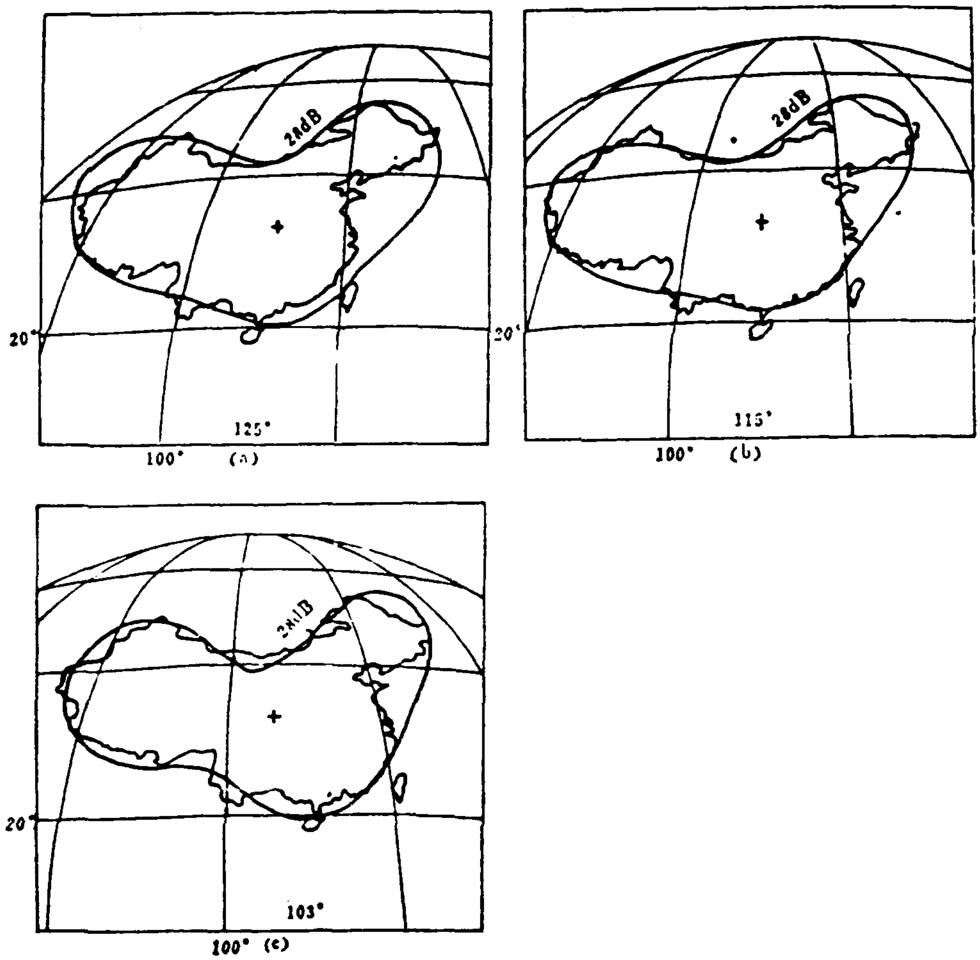

Fig.3 Wave Packet or Beam Coverage Diagrams for Three Fixed Point Locations at 125°, 115°, and 103° (28dB Is Directional Coefficient)

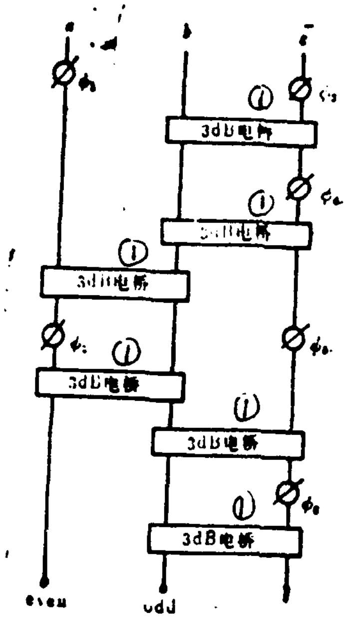

Fig.4 A Block Diagram of 2-3 Odd-Even Synthesizers (1) Electrical Bridge

In our design, we selected for use 2-3 odd-even synthesizers as shown in Fig.4. When we require the three output powers to be according to a certain ratio, then, those three output terminals achieve a set of phase distributions corresponding to the ratio of power and phase. From when the O terminal inputs, take the b terminal zero phase reference point. In that case, the a phase leads the b phase and the c phase lags the b phase. Moreover, if, from the time of the e terminal input, one takes the b (illegible) terminal to be the zero phase reference point, in that case, the a terminal phase lags the b terminal phase. By contrast, the c terminal phase leads the b terminal. Due to there being three output terminals, it follows that it is necessary to take secondary wave packets or beams and divide them into three sets. In the separation of the sets, one should pay attention to making that set of secondary wave packets or beams that is fed in by the center terminal, which has the reference phase of zero, be located, in the arrangement, in the center position of the whole group of secondary wave packets or beams. Take the two sets of secondary wave packets or beams which have positive phases and those that have negative phases and put them respectively on the two sides. In this way, it is possible to avoid the appearance, between

adjacent wave packets or beams, of situations in which phase differences are relatively large. Table 1 gives the phase distributions attached to the various output terminals of 2-3 odd-even synthesizers when there are several different types of amplitude distribution ratios.

In the Table, ϕ_a, ϕ_b, and ϕ_c respectively represent the phases obtained from the three output terminals a, b, and c.

In a situation with the introduction of the odd-even modality synthesizers, the direction diagrams form up and must, at the same time, carry out optimization of the two modalities in order to obtain, in a dual modality situation, simultaneous optimization and the best possible set of excitation parameters. Using this set of parameters 41

① 幅度比	1 : 1 : 1			1 : 0.9 : 0.8			1 : 0.9177 : 0.8885		
② 奇模	ϕ_a	ϕ_b	ϕ_c	ϕ_a	ϕ_b	ϕ_c	ϕ_a	ϕ_b	ϕ_c
	60°	0°	-60°	70.15°	0°	-46.82°	65.08°	0°	-52.24°
③ 偶模	-60°	0°	60°	-70.15°	0°	46.82°	-65.08°	0°	52.24°

Table 1 2-3 Odd-Even Synthesizer Output Amplitude Ratios and Output Phase Distributions (1) Amplitude Ratio (2) Odd Modality (3) Even Modality

of excitation for the synthesis direction diagrams obtained from the various feed sources supplied to them, when the two modalities are operating, in all cases, one achieves optimization. One should point out that, if one compares single modality operating state optimization coverage diagrams and dual modality operating state optimized coverage diagrams (Here, except for the operating modalities being different, the other conditions, in all cases, were the same.) it is possible to see that the dual modality operating coverage characteristics must, as compared to single modality coverage characteristics, differ from them. Generally speaking, they will be lower by 0.5 dB [3]. The reason for this is very clear. Due to the fact that one has opted for the use of odd-even synthesizers, one achieves, in the whole wave

packet or beam formed grid network, the addition of a fixed phase which was not hoped for. Moreover, this phase also follows differences in the operating modalities and changes. In this way, it makes us adjust and optimize the feed source excitation parameters, and, when we do, the constraining conditions are increased. If we, in order to cause these channels, when they are operating, to have antenna wave beam characteristics which are superior to other channels, are able to independently, for a certain modality or certain channels, carry out optimization, the results cause the other modality or channel functions to go down somewhat more.

III. SECOND SATELLITE WAVE BEAM COVERAGE DESIGN RESULTS

In terms of engineering, there are always times when, during the design of a first satellite, due to factors in various areas, it is not possible to simultaneously look after the coverage diagram for the fixed location point of a second satellite. As a result of this, during the design of the first satellite, it is only necessary to consider, under one type of conditions, the carrying out of design optimization. From the design results, one carries out hardware processing. In this way, the second satellite, when it takes over the design, will then be subject to very great limitations. The causes which produce these limitations are that the second satellite, in terms of hardware, has, in a good number of cases, opted for the use of hardware produced on the basis of the first satellite. The limitations this produces manifest themselves in the several areas below (besides the problems described in sections I. and II.):

-The arrangement of feed source arrays are limited by the structures, and it is only possible to select the array and the dimensions of the first satellite. This causes us to be unable to carry out optimization selections on secondary wave packet or beam arrays.

-The selection of polarization direction must be in line with the results from the first satellite. This limitation, in actuality, also reflects the inability to redo selections of secondary wave packet or beam arrays.

-Other geometrical dimensions and weights must, in all cases, be in line with those of the first satellite.

With these types of presuppositions, it is possible to adjust parameters and only have feed source excitation coefficients as a single quantity. Fig.5(a) gives only the secondary wave packet or beam arrangement diagram for after the optimized secondary wave packet or beam arrangement for the service area of the fixed point location at 125° is moved to 103°. From the diagram, it can be clearly seen that it is very unreasonable. Under these conditions, the coverage diagram which is obtained for adjusted excitation amplitude and phase is shown in Fig.5(b). At 125°, according to this type of secondary wave packet or beam arrangement, it is possible to arrive at 90% of the area satisfying the target requirements. However, by contrast, when the fixed point location is at 103°, it is only possible to have 84% of the service area satisfy the coverage target requirements. From this example, one can clearly see that, if the first satellite has optimization carried out on its design for one fixed point location, the second satellite, when it takes over operations onboard at a different orbital location, will produce a number of differences.

IV. CONCLUSIONS

To summarize what we have said above, it is possible to see that, in the C wave band, in high capacity communications satellite antenna subsystems, the technology which we have opted for the use of is equal to the technology which has been chosen for use in the domestic communications satellites of various countries in the world at the present time. However, in design, the problems which have been encountered and the factors which require being simultaneously looked at, in a good number of areas, exceed the current domestic communications satellites of foreign countries. Because of this, it is necessary to go through definite efforts. Only then is it possible
to take antenna subsystems and smoothly and successfully manufacture 42
them. This article only involves problems in the area of electrical design. In measurements, structures, temperature control, and various other similar areas, it is also necessary to make equally great efforts. Only then is it possible to guarantee the whole antenna

system's successful test manufacture.

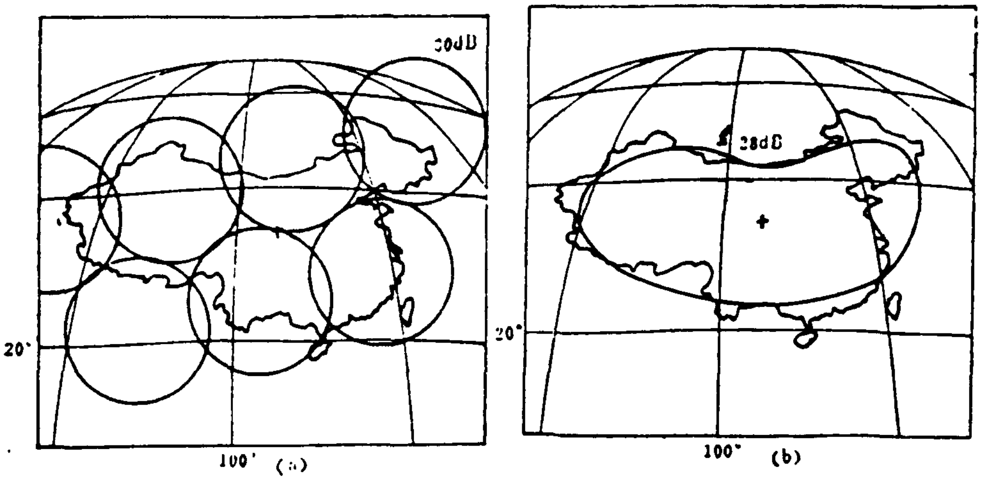

Fig.5 103° Second Satellite Secondary Wave Packet or Beam Arrangement and Wave Beam Coverage Diagrams (28 dB Is Directional Coefficient)

All this article's computational data was worked out by the use of the "Shaped Wave Beam Antenna Computer Assisted Design Software System".

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